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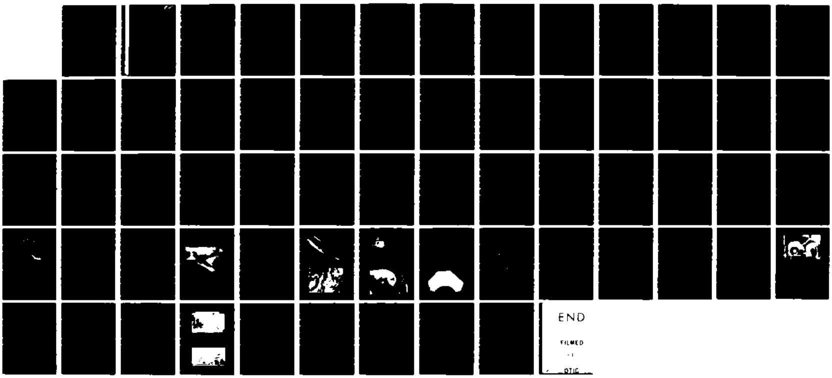

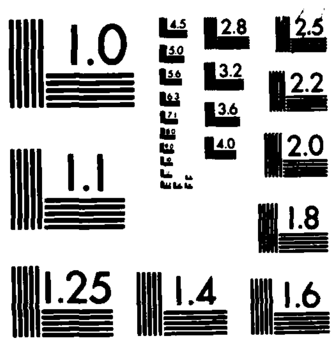

MICROCOPY RESOLUTION TEST CHART
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AUSTRALIA

DEPARTMENT OF DEFENCE SUPPORT

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORIES

MELBOURNE, VICTORIA

Structures Technical Memorandum 359

A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1981 TO MARCH 1983

Edited by

G.S. JOST

Approved for Public Release

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A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1981 TO MARCH 1983

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SUMMARY

This document was prepared for presentation to the 18th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Toulouse, France on May 30 and 31, 1983. It is being distributed within Australia as an ARL Technical Memorandum.

A summary is presented of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical Research Laboratories, the Department of Aviation, the Royal Australian Air Force and the Australian aircraft industry. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage repair and refurbishment and fatigue life monitoring and assessment.

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CONTENTS

9.1 INTRODUCTION

The trend, noted in the previous Review[1], in which increasing fatigue effort is being directed towards the solution of the more urgent shorter term tasks, has been maintained during the past biennium. Competition for limited resources between shorter term projects of more clearcut benefit but limited generality, and longer term projects of less certain outcome but greater potential, has almost invariably resulted in the predominance of the former. Thus, the contributions to the Review reflect, in general, the more applied aspects of fatigue in military and civil aircraft, the evaluation of service fatigue crack growth and life, non-destructive inspection and the repair and refurbishment of fatigue-damaged structure.

This Review has been made possible by the co-operation of the author's colleagues at the Aeronautical Research Laboratories, the Department of Aviation, the Royal Australian Air Force and in the Australian aircraft industry, and their contributions to the Review are gratefully acknowledged. Unless otherwise stated, the topics discussed refer to work carried out at the Aeronautical Research Laboratories; other information has been provided officially by the sources indicated. The names of the various contributing organisations have been abbreviated as follows:

AAC	Australian Aircraft Consortium, Melbourne
ARL	Aeronautical Research Laboratories, Melbourne
DOA	Department of Aviation, Melbourne and Canberra
GAF	Government Aircraft Factories, Melbourne
RAAF	Royal Australian Air Force, Canberra
RMIT	Royal Melbourne Institute of Technology, Melbourne.

9.2 FATIGUE OF MILITARY AIRCRAFT STRUCTURES

9.2.1 Mirage IIIO (ARL, RAAF)

The fatigue test of a complete Mirage airframe has continued at the Swiss Federal Aircraft Factory (F+W) during the past two years. For much of this period the emphasis has concentrated on the validation of the wing main spar rework (Section 9.4.1). This rework has been applied to the fatigue test wings and is being progressively incorporated in wings undergoing refurbishment at the Commonwealth Aircraft Corporation.

In January 1982, a major failure occurred in the flange and associated strap of the main wing load carrying fuselage frame, frame 26, after 8200 test hours. Subsequent inspection revealed cracking at other locations in the same frame. Repair and rework of these cracks enabled the test to continue until 10400 test hours at which stage it became necessary to replace the lower structural strap between the two frame 26 subframes; at the same time the right-hand wing, being close to failure, was replaced by a further refurbished RAAF wing. The test is continuing.

In reponse to the above events ARL has embarked on a program to extend the safety-by-inspection procedures to include frame 26 as well as the wings of all Mirage aircraft. A Structural Integrity Plan for other areas of the Mirage structure assessed as being potentially fatigue critical is also being undertaken. Components of these analyses include:

(a) a program of specimen testing to generate data both to calibrate a crack growth rate computer model and to assess that model by comparison with experimental crack growth data from representative specimens;
(b) fractographic analyses of components from the full-scale fatigue test and from service aircraft;
(c) generation of further flight strain data both from research flights and from carriage of small digital recorders in squadron aircraft;
(d) surveys of selected high-life Mirage fuselages in all potentially fatigue critical areas, inspection of specific areas in all fleet aircraft;
(e) analysis of cracking reported from the periodic inspection of rear flange bolt holes in the wing main spars of RAAF aircraft; and
(f) development of procedures and assessment of the resolution of NDI techniques appropriate for inspection of fuselage frames.

9.2.2 GAF Nomad (GAF, ARL)

The Nomad is an unpressurised, twin turbo-prop aircraft designed and built by the Government Aircraft Factories. It is designed as a utility aircraft with STOL capabilities for operation from short, unprepared airfields. Nomad aircraft are employed in military, civil, geophysical, medical, surveillance and amphibious roles. To date, a total of 135 have been delivered, of the estimated total production run of 170.

To aid certification of the aircraft, a structural fatigue test on a Nomad airframe began in August 1976[2]. The test is based upon a mean cruise weight of 4180 kg [9200 lb] and a one hour flight profile. Eighty per cent of the total landings are based on operation from prepared airstrips. Up to the end of 1982, a total of 137 986 simulated flights have been achieved, at an overall rate of 736 flights per week. Of the 132 different cracks identified to date, 48 have occurred in the main wing, 41 in the stub wing and 43 in the centre fuselage structure. Most of these cracks are minor, and some are considered non-representative and attributable to rig loading.

The fatigue cracks and failures of significance are listed, chronologically, in Table 9.1. Of these, there has been one strut failure, and one detected crack in the front spar of the main wing: the remaining six cracks and failures have all occurred in the stub wing structure. Fig. 9.1 shows the locations of these fatigue critical regions in the stub wing; Fig. 9.2 shows details of the failure which halted the test for over 12 months during 1981/82 pending repair.

The test provides, of course, an ideal opportunity to develop and evaluate repair schemes. Two major repairs have been carried out, each of which has been highly successful. The first was the replacement of the 5.1 mm thick lower wing skin panel in the region of the strut attachment with one of 8.1 mm thickness incorporating increased strut fitting cutout radii. The new skin has survived 25 000 flights without cracking whereas the first crack in the original skin had been detected at 4 500 flights.

A repair was also made to the failed stub-wing front spar lower cap, of which crack growth details are shown in Fig. 9.2[b]. An external bridging strap was designed according to the stress severity factor concept[3] and fitted across the failed region. The repair[4] thus allowed the test to continue with the original spar. This, in turn, allowed subsequent fatigue critical regions in the spar to show up as listed in Table 9.1. The spar has now become so damaged that replacement will be necessary, but the strap repair is still intact after more than 25 000 flights.

Having reached nearly 138 000 flights without either a main wing failure [a function mainly of air loading] or a stub wing rear spar failure [a function mainly of ground loading] the future of the Nomad fatigue test is presently under review. Should the test continue, it is proposed that a new stub wing incorporating the present rear spar be fitted. Thus, structure sensitive to air and ground loading which has already survived the life of the test to date will be retained to reveal its fatigue critical regions in due course.

9.2.3 CT4-A Air Trainer (ARL, RAAF, RMIT)

The CT4-A Air Trainer is a two-place piston engined fully aerobatic trainer aircraft of 1070 kg all-up weight produced in New Zealand for the Royal Australian Air Force.

Since the last report[1] the fatigue test rig has been completely assembled. Other major effort has been directed to:

(a) commissioning of hydraulic and control systems,

(b) analysis of flight trial results, and

(c) the creation of simulated flight load sequences.

In a separate investigation an RMIT fourth year student project investigated

(d) the observed service cracking of corner welds in a CT4-A shear diaphragm[5].

The project at (d) above was noteworthy in achieving a close estimate of observed life by classical means using materials data that was entirely interpolated from handbook values with stress increased by the product of two K_t values. These uncertainties were compensated by almost exact exceedance data and finite element stressing of the wing to the level of boom elements and matching full-web panels with shear stiffness only. Life to cracking was estimated as 2000 hours compared with the observed average of 1831 hours.

The incorporation of service modifications to the fatigue test airframe will normally be made as they are promulgated. For some modifications this has meant that the test article was the first airframe to be modified. The questions involved in such decisions may deserve further attention. A tear-down inspection of the fleet leader has revealed no fatigue problem to date.

The magnitudes of assigned wing load vectors for bending and torsion use least squares fitting to selected gauge strains. For the empennage this strategy failed because the superimposed vibrations allowed a huge variety of bending moment distributions whose magnitudes appear to dominate the fatigue damage process. This was the subject of an investigation[6] for tail bending which disclosed that CT4-A tail and fin were considerably affected by inertia loads from mass balance and the aerodynamic tip loads from full chord horn balances. However the results allowed the fitting of root bending moments and a suitably arranged placement of one-dimensional tail loads. There is some evidence of asymmetry in tail loading.

This type of tail load analysis is, in fact, a case of linear least squares applied to strain readings but with different basis vectors; for coefficients of a parabolic bending moment distribution a tangent construction[6] produces the root shear and bending moment. It is proposed to extend this approach to the combined estimation of empennage and fuselage loads in order to further develop the least squares technique. For any large structural test, it is apparent that there is a relation between the number and type of designated fatigue critical regions, the number of separate loading actions (shear, bending, torsion) and the number of linearly independent load systems.

For the load sequence it is proposed to use the procedure developed for Macchi spars in which flight test sequences are combined to reproduce the service load spectrum. For the test article the service exceedance constraints may need to correspond to each type of mission. The total return period is 5760 flying hours based on repeated 60 hour "training courses" with three levels of rare load additions. These will be stored on the disk of a PDP-1144 computer with overall control of the normal running and rig start-up together with on-line data analysis and automatic calibration of strain responses.

So far, 30 flights have been used in creating tentative load sequences. These will have the larger jack load vectors removed to be used as rarer loads. The lower loads will be truncated to meet the time schedule for the test. The remaining flights relate to high frequency recording, undercarriage and turbulence loads.

9.2.4 Canberra B Mk 20 (ARL, RAAF)

Work on the feasibility of operating Caberra aircraft of the RAAF on a safety-by-inspection basis beyond the nominal safe life has now been completed[7]. As reported in the last Review, all analytical and finite element assessments of crack growth and residual strength in the fatigue critical regions, the lower centre section forging wing attachment lugs, confirmed the practicality of this proposal.

Since that time, crack growth and residual strength data have been obtained on simple specimens manufactured from service life-expired lugs, Fig. 9.3, tested under a random sequence conforming to the current service loading spectrum. (This same sequence had been used earlier in cycle by cycle crack growth prediction). The experimental crack growth rate was found to be very much less than that estimated, and consequently inspection intervals, previously found to be acceptable, now became very satisfactory indeed.

Although this investigation has thus been brought to a satisfactory conclusion. it was decided, on other technical and maintenance grounds, that the RAAF's Canberra fleet should be phased out, and this was effected during 1982.

9.2.5 F-111 Failure of Ultra-High Strength Steel Components (ARL, RAAF)

Low material fracture toughness and high stresses in aircraft structural components implies small critical crack sizes. Where there is also some possibility of manufacturing defects and subsequent fatigue crack development, there is generally a need for regular NDI to ensure continuing structural integrity. For some fatigue critical areas, accessibility is inadequate for conventional NDI, and proof loading (cold proof loading, with reduced fracture toughness and increased yield strength) may be required. The F111 is an aircraft to which all these matters are relevant.

Cold proof loading tests were first conducted on F111 aircraft in 1970. Up to September 1981 about 1500 F111 ultra-high-strength steel wing pivot fitting tests had been performed before a failure occurred. The catastrophic brittle fracture of a fitting on an RAAF F111A at that time revealed a hitherto unsuspected fatigue critical area in that component.

It was established that the suspect region was accessible for conventional NDI, although with some attendant difficulties. In November 1982 a second wing pivot fitting, on an RAAF F111C, failed catastrophically during proof testing, with the fracture-initiating crack in exactly the same region as that in the first fitting to fail. The second fitting had undergone conventional NDI prior to the proof test.

Of particular interest in both cases was the form and size of the fatigue crack which caused failure. For each failure initiation occurred at a large number (approximately 20) of sites across a surface about 6 mm long. In the first case the initiation sites were distributed more or less uniformly right across the surface, although not at the corners, Fig. 9.4(a). In the second case the sites were grouped within a 1.2 mm length, and nearer to one corner of the section, Fig. 9.4(b).

By the time of the fina[illegible], in the first case, a number of "surface" cracks had developed in a li[illegible]ar array, across almost the whole section, Fig. 9.4(a). In the fracture mechanics analysis of that failure, the fracture-initiating crack was considered both as a particular individual "surface" crack [the segment of maximum depth 0.64 mm in Fig. 9.4(a)] and as a "surface" crack of that depth but of length corresponding to the full crack array (approximately 5 mm). It was also considered as an "edge" crack, of the same depth, extending right across the section. The latter crack form implied, of course, a lower effective stress, and in turn lower stress concentration, residual stress and other (geometric) effects.

In the second failure the fatigue crack developed principally about two centres, Fig. 9.4(b), with early joining of the two major cracks and development from there on into a single "corner" crack. The maximum depth of this crack at failure was 2.2 mm.

For the first failure, with the array of surface cracks (or a single surface crack or an edge crack), the nominal failure stress was 235 MPa (34 ksi) and, from fracture mechanics considerations, the actual failure stress was between 345 MPa (50 ksi) and 620 MPa (90 ksi). For the second failure the nominal stress was 275 MPa (40 ksi) and the actual stress 690 MPa (100 ksi) or more. The source of these differences between nominal and actual stress values is being investigated.

9.2.6 RAAF Basic Pilot Training Aircraft (AAC)

The Australian designed BPTA will ultimately form part of a revised RAAF pilot training system, which replaces the CT-4 aeroplane, and which will remain in use at least until the year 2008.

The project management and initial design of the BPTA is being carried out by the Australian Aircraft Consortium (AAC), a private company formed for this purpose by Australia's three largest aircraft organisations - the Government Aircraft Factories, Commonwealth Aircraft Corporation and Hawker de Havilland, Australia. The BPTA is typical of contemporary basic training aircraft philosophy in being of comparatively high performance. This allows a higher proportion of the 100 hour total training syllabus to be carried out on a cost-effective turbo prop before progressing to the higher cost advanced jet trainer.

The RAAF has specified that:

1. the aircraft be capable of susta ning a 2.5g turn at 10 000 feet;
2. the take off and landing distances be less than 500 metres over a 15 m (50 feet) obstacle;

3. low pressure tyres be fitted to provide an aircraft capable of operating from grass or unprepared surfaces;
4. MIL-F-8785B be used for the criterion for static and dynamic stability manoeuvring, stalling and spinning;
5. the airframe have a 20 year life of type for an 8000 hour service life to a specified fatigue spectrum considerably in excess of MIL-A-008866B for basic trainer aircraft;
6. the cockpit accommodate side by side instructor and student with provision for an additional person in the rear; and
7. with manoeuvre flight limit load factors of +7g to -3g.

The resulting configuration, Fig. 9.5, is a prototype aircraft with an AUW of 2000 kg, span of 11 m, length of 10.1 m, reference wing area of 20 m^2, powered by a Pratt and Whitney PT6A-25D turboprop engine, flat rated to 518 KW, and is of all metal semi monocoque construction.

Structural design is to the requirements of MIL STD 1530A as amended by the RAAF to take into account the type of aircraft. This involves the application to the design of damage tolerance and durability assessment procedures. The Structural Integrity Plan for the BPTA is broken down into 4 tasks:

1. Design Information

The design information task encompasses those efforts required to apply the existing theoretical, experimental applied research and operational experience to the specific criteria for materials selection and structural design for the aircraft, such that this task ensures that the appropriate criteria and planned usage are applied to the BPTA design so that its operational requirements will be met.

Materials selected for use in the BPTA incorporate 7475, 7010 and 7050 series aluminium alloys for fittings, extrusions etc and HP 9.4.20 steel for wing/fuselage and tailplane/fuselage fittings.

2. Design Analyses and Development Tests

The objectives of the design analyses and development task will be to determine the environments in which the airframe would be expected to operate and to perform preliminary analyses and tests based on these environments to design and size the airframe to meet the required strength, damage tolerance and durability requirements.

Coupon testing for crack propagation data not available from the literature for the steel and aluminium alloys to be used has commenced. Selected components will be subjected to flight-by-flight sequence loads.

3. Full Scale Testing

The structural certification test programme consists of static and durability tests of the airframe and static and fatigue tests of the gear. Damage tolerance test requirements will be met by fractographic examination and crack growth analysis of any structural failure that occurs in the full scale durability test specimen.

The full scale airframe durability test has been scheduled to reach two design service lifetimes before assembly of the first series production aircraft. The durability test is then required to demonstrate an economic life of four design service lifetimes.

4. Fleet Management Data Package

The purpose of this task is to provide information to the RAAF on inspection and repair criteria together with strength summaries, such that the RAAF can maintain the strength, rigidity, damage tolerance and durability of the aircraft fleet. Further experience with other aircraft types has shown that the actual usage of RAAF aircraft may differ significantly from the assumed design usage and this task would identify further RAAF actions required to obtain relevant data.

Initial conceptual design is complete and detail design is now being carried out by the three particpating companies. The prototype programme involves the production of four airframes, numbers 1 and 4 flying and numbers 2 and 3 for static testing and durability testing respectively. The first flight is due in February 1985 with the award of type certificate by mid 1987.

9.3 FATIGUE IN CIVIL AIRCRAFT

9.3.1 Lockheed L188 Rear Pressure Bulkhead Fatigue (DOA)

An outline of the difficulties which can be associated with operating "geriatric" aircraft types such as the Lockheed L188 Electra, was given in the previous Review[1], and detailed the experience of discovering previously unknown and potentially catastrophic fatigue cracks in the BL65 wing root splice area. Since then yet another previously unknown problem has manifested itself; this time in the rear pressure bulkhead.

After the flight crew of the freighter aircraft reported difficulties with the pressurisation system, an investigation of the equipment bay aft of the rear pressure bulkhead revealed obvious trauma to the thermal insulation blanket. Visual inspection of the bulkhead revealed a 54 cm circumferential crack in the pressure web on the left hand side of the aircraft, with a corresponding 16 cm crack in the underlying doubler, and a 15 cm crack in the web on the right hand side. The aircraft total time in service was 49 470 hours, 38 469 landings. The cracked sections of the bulkhead are shown in Fig. 9.7(a).

The L188 rear pressure bulkhead consists of a 0.8 mm 2024-T4 dished pressure web attached to a ring frame, formed by two back-to-back angles, Fig. 9.6. The web is supported by radial stiffeners on the aft face, and circular tear stoppers on the forward (pressure) face.

Optical and SEM fractography, Fig. 9.7(b), of the 54 cm web crack showed it to be a multiple origin fatigue crack consisting of a number of small cracks initiating at rivet holes through the web and doubler[8]. These small cracks were from 3 to 10 mm in length and had joined by overload to a total length of about 10 cm ("zip fastener" effect). The remainder of the crack was caused by overload. The fatigue crack growth had occurred over a period in excess of 5 000 pressure cycles. The 16 cm doubler crack and the 15 cm web crack were also confirmed as fatigue cracks resulting from cabin pressurisation.

This occurrence, with its rapid crack growth from 10 cm to 54 cm, is in some respects reminiscent of the Vanguard accident in Belgium in 1971[9], but fortunately without the secondary effect and tragic consequences of that accident.

The lessons from this occurrence are again the same. Although the bulkhead was subject to general visual inspection at 4 000 hour intervals, specific and detailed inspection techniques and criteria were not given. In this instance the area had last been inspected some 3 850 hours prior to the incident, at which time cracks were present and were detectable. It is perhaps significant that both the BL65 wing splice area previously cited[1], and the rear pressure bulkhead, were thought at the time not to be of sufficient concern to merit inclusion in a proposed Supplemental Inspection Document program. This raises the question of whether other areas have been overlooked, and whether the SID programs for other "geriatric" aircraft types have covered all possibilities.

9.3.2 Beechcraft Wing Fatigue (DOA)

The Beech 50 Twin Bonanza, first produced in 1951, had a good wing structure with low stress levels and high margins of safety. The same wing spar design was then used on the whole of the Queen Air, King Air and later on the Swearingen Merlin II aircraft. The airworthiness standards to which these aircraft were designed did not include a wing fatigue requirement, nor was one imposed despite an eventual doubling of gross weight and very large increases in power and speed. However, the situation changed somewhat in the late 1960s when Beech started a full scale wing fatigue test program in order to gain FAR Part 135 Air Taxi certification for the Model 99 airliner.

A number of fatigue test results of main spar lower cap and outer attachment fittings became available in 1970 and these formed the basis of Australian Airworthiness Directives covering the full range of Beech aircraft. Generally, the Australian approach was to place a retirement life on the wing centre-section and outer wing main spar lower caps. This followed from the identification of 10 possible failure locations in the wing and the difficulty of carrying out reliable inspections.

The only exception to the above policy concerns the outer wing main spar lower attachment 'bath tub' fitting which, on several models, has a lower life than the remainder of the wing spar. Test experience showed that the integrity of the fitting was capable of being controlled by inspection and Airworthiness Directives were promulgated as appropriate. Fig. 9.8(a) shows one of these fittings. Fig. 9.8(b) shows a typical dye penetrant crack indication found on a fitting during fatigue testing. It is significant that the location differs from that illustrated in the applicable Beech Service Instruction.

Early in the period under review the outer wing fittings on an Australian operated Beech 65-A80 aircraft were inspected using the dye penetrant technique. The aircraft concerned had over 6000 hours time in service. A small crack indication was found in one fitting on the upper surface at the blend-out of the washer face radius. Not only was the crack indication not reported to the Department of Aviation, in contravention of Australian Air Navigation Regulations, but an attempt was made to dress out the indication. This process was repeated 100 hours later although this time the Department was informed after a further attempted "removal of the indication".

The affected part was removed from the aircraft and the fitting was subjected to metallurgical and NDI examination[10]. No crack indication was evident by dye penetrant examination but following positve eddy current and ultrasonic indications the fitting was broken open. A massive fatigue crack was discovered running 29 mm around the radius and extending almost through the complete wall thickness. Fig. 9.9(a) illustrates the fracture section and Fig.9.9(b) is a magnified view showing the fatigue nature of the cracking.

Fractographic examination of the fracture face using a scanning electron microscope indicated that the crack had been growing for some 400 flights and comparison with two failure cases suggested that the wing would most likely have separated from the aircraft within a short time, possibly as few as 90 flights. It must be noted that the manufacturer arranged for an independent fractographic analysis of one face of the fracture. Whilst this work did not specifically disagree with the Australian analysis, it concluded that the nature of the crack propagation did not allow correlation between striation markings and specific loading events.

The following points are relevant:

1. The use of a fluorescent penetrant is recommended in the Airworthiness Directive. This recommendation was made in the knowledge that a high test sensitivity was required. It appears that some operators may have ignored this and used the more easily applied, but less sensitive, red dye materials in ignorance of the fact that, if a doubtful indication is obtained using red dye, it is not possible to recheck using the more sensitive fluorescent materials since the presence of red dye in a crack will quench the fluorescence of the dye. This was emphasised in a subsequent revision of the Airworthiness Directive.

2. In cases where it may be practicable to dress out a crack indication, considerable care must be exercised. In the above case a rotary file was used which not only removed 0.3 mm from the surface but also smeared metal over the crack opening thus preventing further identification of the defect. In such reworks, the minimum of material should be removed by hand-sanding using well-lubricated "wet-and-dry" paper. The aim should be to provide a cutting action. A lightly applied coarse emery is less likely to cause metal to be smeared across the crack opening than is a heavily applied fine emergy. Naturally, very coarse emergy cannot be used, since deep scratches must be avoided. An acceptable compromise appears to be the use of around 240 grit paper.

9.3.3 Aircraft Wheel Fatigue (DOA)

Fatigue cracking of landing wheels is a continuing problem with current narrow and wide bodied transport aircraft and will continue to be a significant airline maintenance cost burden for some time to come. The eddy current test method has been established as a most effective method of inspecting aircraft wheels for fatigue cracks at tyre change periods[2].

The progressive introduction of aircraft tyres to the higher design standards of the American Technical Standard Order C62c, which includes a 7 per cent tyre load margin, has caused the wheel manufacturers to progressively upgrade their wheel designs to match TSO-C62c tyre standards.

9.3.4 Gas Tubine Engine Rotor Failures (DOA, ARL)

The Department of Aviation is becoming increasingly concerned with gas turbine engine rotor failures especially when viewed on a global basis. The incidence of rotor failures in larger turbine engine types overseas has prompted the Department to monitor carefully the operation and maintenance of engines of the same types operated in Australia. The U.S. Federal Aviation Administration's Service Difficulty Reports show that for these engines uncontained rotor failures have averaged approximately 1 per million engine operating hours over the last 7 years.

Two local uncontained failures, one involving power turbine blade shedding from a Canadian Pratt & Whitney PT6-34 and the other, a Garrett TPE 331 third stage turbine rotor together with a close study of the uncontained failures overseas of two General Electric CF6-50 high pressure tubine disks indicate there are still shortcomings in turbine rotor life and integrity prediction techniques.

Some review work on these techniques has been initiated with the Aeronautical Research Laboratories. This is seen as a difficult area of investigation because of the proprietary nature of much of the engine manufacturers' technology.

9.3.5 Fatigue of Fibre Reinforce Plastic (FRP) Gliders (DOA, RMIT)

Australia has a very active gliding movement, with more than 900 gliders and motor gliders on the aircraft register. Some 500 of these are FRP machines. The airworthiness certification and control of gliders has been delegated to the Gliding Federation of Australia, with specialist advice and assistance being provided by the Australian Department of Aviation as necessary. A number of FRP glider types of German manufacture are achieving fatigue life limitation problems because of their extremely high utilisation. The relevant glider design code is largely related to European operating conditions and this has led to a 3 000 hours or 15 years total life criterion.

Many gliders operating in Australia have far greater utilisation than that envisaged by the design code or by the manufacturers, and so a number of gliders are now reaching, or have already exceeded, the 3 000 hours life limitation. The Department of Aviation and the Gliding Federation of Australia have been active in negotiating life extensions for some FRP gliders with the manufacturers and their regulatory authority, but with limited success. The problem is currently being addressed on the basis of Manufacturers' Life Extension Inspection Schedules. However, the data base for the life extensions is usually a combination of the physical inspections of time expired gliders, and the results of cyclic fatigue testing of new components. This is most unsatisfactory from an airworthiness viewpoint, and full scale fatigue testing of representative wings is seen as the most rational way of dealing with the problem in a quantifiable manner.

To this end, arrangements have been made with the Royal Melbourne Institute of Technology to conduct a full scale fatigue test on a 'Janus' glider wing, under sponsorship from the Gliding Federation of Australia and the Department of Aviation. A flight strain survey is to be carried out, and two Janus gliders have been fitted with Fatigue Meters to define the load spectrum. The RMIT's Janus B fibreglass glider, which is used for student instruction, has been instrumented with electric resistance strain gauges at five spanwise stations on the wing. In addition to recording strain outputs from these locations, horizontal and vertical centre of gravity accelerations, pitch rate, height, speed and flap position are recorded on a compact magnetic tape unit developed by RMIT. The glider is being flown in soaring competitions this summer by experienced sailplane pilots with an observer to operate and monitor the recording equipment. To reduce recording time and the corresponding data reduction time the observer selects recording intervals throughout the flight and maintains a flight log. It is proposed to continue the investigation over an extended period, operating the glider both in gliding competitions and in general club flying. Investigation of the ground to air cycle and taxying loads will also be included, and aeroelastic effects will be studied.

The full scale test specimen comprises a complete wing pair. One of these is a brand new wing, the other has been in service for some time and has suffered major accident damage. The test will therefore also help evaluation the effectiveness of major repairs. It is proposed to continue the test to failure or structural distress to establish the mode(s) of failure, and hopefully develop appropriate inspection procedures that will eliminate the necessity for a 'hard time' life limitation.

Concurrently with the flight loads and strain survey, a research programme has begun on fibreglass specimens to provide basic fatigue and residual strength information under characteristic loading sequences. This will provide a basis for the fatigue substantiation of fibreglass gliders in general.

9.3.6 Fatigue of Fibre Reinforced Plastic Helicopter Components (DOA)

During the period under review there have been four separate defects involving cracking of one arm of the FRP main rotor "star" on the only Australian registered Aerospatiale SA365 "Dauphin" helicopter. As far as can be ascertained there has been only one other similar defect found elsewhere.

These small chordwise cracks have been found about mid-way along the arm, in the trailing edge, at times ranging from 71 hours to 459 hours time in service. They were found by visual inspection. They are still under investigation and the cause has not yet been established; indeed there is as yet no confirmation that it is a fatigue problem.

9.4 FATIGUE DAMAGED STRUCTURE: ANALYSIS, REPAIR AND REFURBISHMENT

9.4.1 Mirage Fatigue Life Extension Program (ARL, CAC, RAAF)

9.4.1.1 Rear flange

In the 1981 Australian ICAF Review, and also in a paper presented at the 11th ICAF Symposium[11], details were given of the development of a refurbishment scheme for extending the fatigue life of the lower inboard rear flange of the main spar of Mirage IIIO fighter wings. The type of specimen which was used for the investigation is shown in Fig. 9.10. This investigation has now been completed and the results are summarised in Fig. 9.11. Cold-expansion of the bolt holes (2.7%) was done using the Boeing Split-Sleeve process; the interference-fit adopted for the steel bushes was nominally 0.3%. Although the interference-fit bushing process has been used to successfully extend the life of the RAAF Mirage wings, it cannot be overemphasised that the effectiveness of this process in increasing fatigue life relies to a large extent on the maintenance of the correct bush interferences which, in turn, require careful quality control in the machining of both the hole and the bush and also in the bush insertion.

9.4.1.2 Front flange

As a complementary investigation to that on the rear flange of the Mirage main spar, tests were carried out to explore methods for extending the life at the front flange. These involved specimens of the type shown in Fig. 9.12 and included the following hole treatments:

A. 5 mm straight-shank interference-fit bolts - these had an interference of 0.4% in the holes and corresponded to the original structure design detail;

B. 5 mm straight-shank close-fit (transition-fit) bolts;

C. Cold-expanded holes - nominally 3% using the Boeing Split-Sleeve process, and fitted with 0.250 inch (6.35 mm) clearance-fit bolts;

D. Cold-expanded and bushed - cold-expanded as in (C), but then reamed to 7 mm diameter and type 304 stainless steel bushes with nominally 0.3% interference fitted and standard 5 mm clearance-fit bolts fitted in bushes;

E. Interference-fit bushes - stainless steel interference-fit bushes of 8.15 mm outside diameter, again incorporating 5 mm bolts.

All tests were carried out using the same 100 flight loading sequence as that used for the tests on the rear flange specimens. In this case the +7.5g load corresponded to a gross area stress of 235 MPa. The test results are summarised in Table 9.2, and typical crack growth curves for treatments B and C above are shown in Fig. 9.13.

This investigation has clearly demonstrated the marked superiority of interference-fit bolt compared to transition-fit bolt fastening systems - a ratio of almost 10:1 in average lives. Although the cold-expanded hole and interference-fit bushing systems produce greater net area stresses than those incorporating 5 mm bolts, cold expanding or the installation of interference-fit bushes or a combination of these two techniques provides increased lives compared to the use of transition-fit bolts. In a situation such as the refurbishment of the Mirage IIIO main spar (where the bolt holes must be enlarged to remove fatigue cracks and the requirement for hole-inspection now precludes the use of interference-fit bolts) the use of interference-fit bushes, either alone or in combination with cold-expanded holes, should enable useful extensions in fatigue life to be achieved.

9.4.2 Stress Fields Around Interference-Fitted and Cold-Expanded Holes (ARL)

A review[12] has been made of the relevant problems involved in accurately defining the stress fields surrounding holes (previously cold-expanded or fitted with interference-fit fasteners) in both unloaded and remotely loaded plates. Although various theoretical, numerical and experimental methods have been used to predict or measure the stress fields associated with these fatigue life-enhancement systems, there are significant shortcomings in the models - particularly in the assumptions regarding the elastic-plastic behaviour of the plate material - and there are very few systematic fatigue test data to enable the validity of any predicted behaviour under cyclic loadings to be assessed. It can fairly be said that the demonstrated effectiveness of these fastener systems has not been matched by a corresponding clarity of understanding of the fatigue improvement mechanisms responsible. In view of the increasing use of these fatigue-life-enhancement systems it is clearly desirable that a method be developed whereby processing conditions to provide the maximum fatigue life for any particular structural fatigue loading situation can be precisely specified. A program of theoretical and experimental stress analysis supported by extensive fatigue testing is being planned to meet this objective.

The most critical holes at joints are often those closest to an edge, and in an exploratory project, Jeffery's work[13] on a pressurised hole in a semi-infinite plate has been used to represent a wholly elastic interference-fit bush case. By relating interface pressure to the average interference between hole and bush, excellent agreement between calculated and measured hoop strains was obtained[14], Fig. 9.14. The principal stress field is shown as a contour plot in Fig. 9.15. The study is being extended into the plastic region.

9.4.3 Crack Closure and Overload Effects in Fatigue

A study has been carried out to investigate crack closure and overload effects in fatigue[15]. In the tests edge-cracked aluminium alloy specimens were subjected to simple overload sequences, whilst the crack-tip hysteresis behaviour was monitored by a surface-mounted displacement gauge (0.5 mm behind the crack tip). Detailed changes in crack growth rate before and after the various overload sequences were compared with the measured hysteresis loops. The crack-opening stress was found to decrease in the stage immediately following a short overload sequence (up to ten cycles); however, no corresponding increase in fatigue crack growth rate was observed. In general, no direct correlation between the measured hysteresis behaviour and the crack growth rate was found. It was concluded that the crack growth rate under varying load can be influenced by a local crack-closure condition which is not simply related to the crack-opening stress.

9.4.4 Fatigue Cracking and Microstructure in Ti-8Al-1Mo-1V Alloy

In the titanium alloy Ti-8Al-1Mo-1V, crack propagation rates have been measured for a wide range of microstructures obtained by different heat treatment procedures. These microstructures include equiaxed alpha phase grains plus beta phase particles, equiaxed grains of alpha phase plus grains of transformed beta phase and grains of transformed beta phase. The scale of features represented in these microstructures ranges from 10 to 200 micrometres, but the rate of crack propagation, for a constant ΔK, remains similar for all microstructures over the range $\Delta K=20$ to 54 $MPa\sqrt{m}$.

At lower stress intensities ($\Delta K=4$ to 8 $MPa\sqrt{m}$ corresponding to da/dn $\sim 10^{-10}$ to 10^{-8} m/cycle), the crack growth rates become much more microstructure sensitive. Cracks in the specimens of duplex annealed, equiaxed alpha grains plus beta particles propagate at up to 7 times the rates of those in specimens of alpha phase plus intergranular beta phase.

Crack path morphology indicates a marked increase in crack branching as ΔK decreases. The scale of the branching is microstructure dependent: larger branching occurs in microstructures comprising Widmannstatten transformed beta, and wide spread small scale branching occurs in alpha plus transformed beta and equiaxed alpha plus beta particles.

9.4.5 Statistical Variation in Fracture Toughness of Titanium Alloys (DOA)

A small project has been initiated with the RMIT to produce fracture toughness probability density functions from pooled published test data on Ti-6Al-4V. The exercise was primarily initiated to determine if the data best fit one or other of the five distribution functions:

1. Normal distribution function
2. Log-normal distribution function, 2 parameter
3. Log-normal distribution function, 3 parameter
4. Weibull distribution function, 2 parameter
5. Weibull distribution function, 3 parameter.

The data plots displayed an overall "poorness of fit", the worst being with the Normal distribution, and the best being obtained from the Log-normal and Weibull plots. The fits were tested using appropriate probability plots using linear transformation techniques for the various probability density functions.

The exercise revealed a very wide scatter in the data and suggests at least qualitatively that inadequacies in the materials data base may be a possible explanation for the differences in fatigue and crack growth behaviour of large Ti-6Al-4V helicopter components and those predicted from small scale specimen tests.

9.4.6 Fracture Mechanics

With the increasing interest in the adoption of the J-integral fracture criterion, an investigation has been undertaken of its efficient numerical calculation[16].

9.4.7 In-Service Crack Growth Model for Mirage

As noted in the last Review, cracks have been detected in the rear flange bolt holes of the main spars of many RAAF Mirage aircraft. Using information from the first fleet-wide inspection of the two most inboard bolt holes, a statistical analysis of the crack data was carried out with a view to developing a crack growth model for use in fatigue life assessment. Statistical tests suggested that crack growth behaviour at the two holes is similar, but were inconclusive as regards the significance of an apparent difference between port and starboard wings. An important feature of the data was the very poor correlation between cracking severity and both flying bours and loading severity (the latter being available for individual wings from fatigue meter data).

The inspection data alone were insufficient to develop a consistent model of crack growth in service. However, when the data were interpreted in conjunction with a limited amount of crack growth data obtained by fracto-graphy from a small crack in a crashed aircraft, it was found possible to obtain a model consistent with all the available information from the fleet[17]. The implication of this model is that, in the absence of any refurbishment programme for the spars, the mean and safe service lives for the wing are considerably shorter than that indicated by fatigue testing.

9.4.8 Crack Patching - Theoretical Analysis

Theoretical work on crack-patching has elucidated the key concepts required for assessing the efficiency of bonded repairs to cracked plates. The problem is conveniently viewed in two stages. First, the redistribution of load which would be produced by the bonded patch in an un-cracked plate must be assessed. This can conveniently be done by treating the reinforced region as an inclusion of higher stiffness than the surrounding plate. Analytical results have been derived[18] which show, in particular, the effect of the aspect ratio of the patch. The second stage is to cut in the crack. The essential concept here is that an upper bound for the relative displacement of the crack faces can be derived by considering the displacement in an overlap joint. This leads to upper bounds for the crack extension force, the maximum shear stress in the adhesive layer and the maximum fibre-stress in the reinforcing patch[19]. Thus, analytical formulae are available which allow a quick preliminary design assessment of proposed repair schemes.

9.4.9 Boron Fibre Reinforced Plastic (BFRP) Crack-Patching

The crack-patching procedure, pioneered by ARL for fatigue cracked aluminium alloys (described in Australian ICAF Reviews from 1977) has been further developed and evaluated, and considerable service experience has been gained with some repairs[20].

Basically, the repair procedure involves the use of unidirectional boron fibre reinforced plastic (BFRP) patches which are adhesively bonded over the cracked region. The repairs are made with the fibres spanning the crack. These restrict the opening of the crack under load and thereby reduce stress-intensity, preventing or slowing further crack growth. Adhesive bonding provides very efficient load transfer into the patch from the cracked component and eliminates the need for additional fastener holes, which could act as stress raisers. The advantages of using BFRP for the patch material include high directional stiffness, good resistance to damage by cyclic loading and corrosion, and excellent formability. BFRP was chosen over the cheaper carbon fibre reinforced plastic because of its better combination of fatigue strength and stiffness, its higher thermal expansion and its low electrical conductivity. The adhesive chosen for most applications is an epoxy-nitrile structural film adhesive, curing at about 110°C. Where possible, the surface of the aluminium component is treated prior to bonding by the PANTA procedure developed by Boeing; if this treatment is not possible the surface is thoroughly blasted with alumina grit.

9.4.9.1 Experience with repairs to Mirage III aircraft

Fatigue cracks in lower wing skins of RAAF Mirage III aircraft have been repaired using the Boron/Epoxy procedure[21,22]. Patches were also applied to uncracked wings as a precautionary measure. This application program was completed in October 1981. In-service monitoring of cracks is being performed using an eddy current inspection procedure. Fig. 9.16 shows the repair applied to the Mirage under fatigue test in Switzerland.

To date, seven reports of post patch crack growth in service have been received. One wing skin was replaced as a result of the cracking, and another wing skin was replaced as part of a refurbishment program. The remaining five are still in service, and four of these have shown no evidence of crack growth after the initial report. The arrest of these cracks has persisted for up to 400 hours of service. The remaining wing skins where crack propagation has occurred has been repatched after splitting of the patch was detected due apparently to inadequate pressurisation of the adhesive during repair. A supplementary repair was applied, in conjunction with the replacement boron repair.

A boron patch was also applied to the same region of a wing on the Mirage undergoing full scale fatigue testing in Switzerland. The patch was applied to a crack which was above the allowable limits of repairability for a fleet aircraft; further, there was evidence of a malfunction during the application procedure. This resulted in continued propagation of the crack in the forward direction at an early stage of testing. However, the patch system exhibited arrest of the crack in the other direction for more than 3 000 test hours.

Concurrently with the repair to the fuel drain point on the fatigue test aircraft, a further repair was applied to cracks initiating from an adjacent fastener hole employed as one of the points of attachment for the wing to fuselage fairing; this is shown in Fig. 9.16, right. In this case, the repair withstood in excess of 4 000 test hours and was still serviceable at the conclusion of testing of the wing. Techniques for implementation have been developed to the stage where fleet incorporation of the repair is now possible; some field repairs have recently been performed. The development and subsequent implementation of this repair form part of a program to transfer crack patching technology to the Australian aircraft industry.

9.4.9.2 Design studies

Two design approaches to crack-patching have been developed. The first is based on a detailed three dimensional finite element model. This is a development of the two dimensional procedure used in the development of the repairs to Mirage.

The three dimensional model uses 3-D isoparametric elements to model each component (plate, adhesive layer and patch). Reduced integration is used throughout and in the sheet the nodes nearest the crack front are moved to quarter points in order to simulate the required $\sqrt{r}$ singularity. To date, the numerical results show excellent agreement with their corresponding experimentally measured values. This work has also confirmed the accuracy of the 2-D design charts and formulae previously developed at ARL.

A hybrid finite element has also been developed to investigate the use of unbalanced patches ($0_7/+45/-45_2/+45$). It has been shown that by using unbalanced patches the effects of neutral axis shift, i.e. secondary bending, can be significantly reduced[23].

The second design approach is analytical, and is therefore useful in allowing a detailed understanding of the influence of the materials parameters and also in providing a simple conservative estimate of K reduction in practical applications where the more comprehensive finite element approach would not be cost effective or sufficiently timely.

It is shown[24] that the reduction in stress-intensity K_r for a centre-cracked plate repaired by bonded reinforcements can be estimated by considering an overlap joint whose section corresponds to a section taken through the cracked repaired plate where the reinforcement covers the crack. This is illustrated in Fig. 9.17(a) and (b). The situation where the patch does not fully cover the plate is allowed for in the analysis. The result of the analysis is that, as Fig. 9.17(c) illustrates, K_r is found to approach an upper bound, K_∞, which implies that the crack appears to have a maximum apparent size, λ, irrespective of its true size, a.

The simplest practical joint relevant to the repair context is the double-overlap joint illustrated in Fig. 9.18, which represents a two sided repair in which patches are bonded to both faces of a cracked plate. The variable which is of greatest relevance to the theory is the relative displacment δ shown in Fig. 9.19, because this is used to derive an upper bound for K_∞.

In order to check experimentally the theoretical estimate of δ based on a one dimensional analysis, some experimental checks were made using the specimen design shown in Fig. 9.18, and employing standard epoxy-nitrile film for the adhesive. A clip gauge is used to measure Δ which then provides an experimental value for δ. The full set of results[24] shows that theory and experiment agree quite well if a shear modulus of about 0.5 GPa is assumed for the adhesive and a shear lag correction is made for the BFRP. For example, in a typical case, the theoretical value of δ was found to be 28 μm and the correct experimental result was 30 μm.

9.5 FATIGUE LOADS, LIFE MONITORING AND ASSESSMENT

9.5.1 Flight Loads Spectra (DOA)

The civil aviation scene has been changed over the last few years by the development and widespread acceptance of a number of twin turboprop, unpressurised, commuter aircraft. As more designers turn to this kind of aeroplane, there is an increasing demand for measured flight loads data, both to enable review of fatigue substantiations for existing aircraft types and to provide input to fatigue tests and analyses for proposed future aircraft.

Four such aircraft operating in Australia have been fitted with Fatigue Meters. The results to date are shown in Fig. 9.20, and the corresponding operational data are given in Table 9.3. The results for all except the DHC-6 are subject to correction for Fatigue Meter calibration, however any resulting effect is usually small.

9.5.2 Magnification Factors in MIL-A-83444

In the U.S. Damage Tolerance Requirements[25], the required minimum residual strength P_{xx} is the load having a return period equal to the product of the inspection interval and a magnification factor M which is intended to allow for between-aircraft variability in loading frequencies. The factor M is itself a function of inspection interval and ranges from 20 to 100. In an attempt to validate the rationale behind these numbers, the between-aircraft loading variability for the RAAF Mirage fleet has been investigated and an assessment has been made of the increased safety arising from the use of such values of M.

The fleet has been in service since 1964. Fatigue meters have been fitted to all aircraft from that time and approximately 240 000 hours of data have been collected from the fleet as a whole. These data give a confident estimate of the variability in loading frequencies up to 6.0g, but at the highest fatigue meter threshold, 8.0g, the paucity of data provide a less confident variability estimate. The frequency at 6.0g was found to have greater variability than those at lower g-levels, and it was assumed that the 6.0g variability is typical of that at high loads generally, both in the Mirage fleet and in other "fighter-type" aircraft fleets operating throughout the world. Subject to this assumption, the preliminary indication is that the values of M recommended in MIL-A-83444 are somewhat conservative[26].

9.5.3 Composite Materials - Damage Tolerance Aspects

Since Australia is in the process of procuring the F/A-18 aircraft which uses a significant amount of graphite/epoxy composite in its airframe, increasing effort is being devoted to consideration of damage tolerance problems for composite structures. A review of the field has been undertaken[27], and a theoretical method has been developed for determining the weakening effect of a delamination on the compression strength of a composite laminate[28]. Whilst this method has still to be fully assessed against test data, the results to date seem promising.

9.5.4 Load Cycle Reconstitution

Most data acquisition systems monitoring aircraft loads delete sequence information for convenience. When flight loads data from such condensed storage systems are used as the basis of fatigue test programs, or for fatigue life prediction, load sequences must be reconstituted artificially.

Many investigations have shown that for simple load sequences applied to specimens of simple geometry, and especially for the single overload case, the order of application of loads has a significant influence on fatigue life. There is speculation, however, as to the relevance of these results for more complex sequences and geometries, such as those pertaining to aircraft in service.

An experimental program has been started to determine, for a given condensed data set, the range of crack propagation lives for several possible cycle reconstitutions from that set. The aim is to discover whether this range is small or large for reconstitutions of flight loads for a military aircraft and for a representative specimen geometry. The general principles of creating crack growth retardations and accelerations are considered in determining schemes of reconstitution; e.g. for retardation, large negative loads should be followed by large positive loads, and a large positive load should be followed by small positive loads.

The basic sequence adopted was obtained from that measured near the wing root on the main spar in the Australian Mirage wing fatigue test[29]. The wing test sequence comprised 500 flights and about 100 000 turning points (about 50 000 cycles or pairs) before repetition. The peaks and troughs of this sequence were rounded to the mid-points of a counting grid resulting in 14 discrete load levels. A range-mean pair count was then made on this sequence giving the values shown in Table 9.4. All sequences reconstituted from this Table must, of course, give the same range-mean pair table entries.

One simple method of reconstitution, and thought, initially, to follow some of the principles of creating crack growth retardation, is to begin with the largest load range and follow with the next smallest and so on, the order within each load range following successively lower mean loads. This is achieved by commencing at the highest peak - lowest trough (1820 to -520 in Table 9.4) and following with blocks from successive diagonals, each diagonal going from lower right to upper left. This sequence is provisionally termed 'retardation'. The reverse sequence was termed 'acceleration'; in this case the order was from the uppermost left block (-340 to -520) along the diagonal to the lowest right block (1640 to 1460), then following in the same pattern along successive diagonals until the maximum peak - minimum trough pairing occurred.

The test piece used represented an area of the rear flange of the Mirage main spar which features a heavy section with bolted connections and load transfer, Fig. 9.21. The few tests which so far have been completed were made at a net-area stress of 35.6 MPa per g, the maximum load in the original wing test sequence being 7.5g. The results are given in Table 9.5, with a single program defined as the application of all load pairs given in the range-mean pair table.

It is clear from these results that either the range of crack propagation lives over all possible reconstitutions is very small or that the principle of reconstituting along diagonals in the range-mean pair table is inadequate. If we assume little scatter, the results above also indicate that reconstitution aimed at producing crack growth retardation in fact did not, or at least it produced less retardation than the 'acceleration' reconstitution.

This latter point is supported by fractographic examinations. Fig. 9.22(a) shows a macrophotograph of the fracture under the 'retardation' sequence, and Fig. 9.22(b) shows the 'acceleration' fracture. Dark areas indicate tensile fracture and light areas indicate striation fatigue. The retardation sequence shows evidence of a good deal more fatigue fracture due to intermediate- and low-value loads than does the acceleration sequence. Nevertheless, on average, it takes only 0.8 of a program under the retardation sequence to produce the same amount of cracking as one program under the acceleration sequence.

The fractographic evidence above, and the test lives, indicate that more than one program should have been considered in arranging the reconstitutions. The end of one program and the beginning of the next for the retardation reconstitution gives an excellent crack growth acceleration sequence, namely, small-amplitude loads followed by large-amplitude loads; and vice versa for the acceleration reconstitution.

A further reconstitution is being arranged, based on the finer principles of crack growth retardations in simple specimens, and hopefully it will maximise fatigue life. It is based upon three considerations. First, the Willenborg model of crack growth retardation following a single overload predicts complete cessation of growth when σ_{max} (overload) $\geqslant 2\ \sigma_{max}$ (subsequent constant amplitude load). Second, delayed retardation commonly occurs, ie. the crack growth rate does not instantly reduce to a minimum after the application of an overload. Third, multiple overloads hasten the delay and reduce the value of the minimum rate of subsequent growth.

To utilize these principles the following sequence will be applied.

1. Begin with the highest load range, 1820 to -520 (one pair only available), and follow this with all load pairs with peaks less than half the maximum (using Willenborg).
2. The 9 pairs of the next highest range (1640 to -520) will then be applied as multiple overloads, following with the row of blocks commencing 920 to 740, working towards the left. This arrangement utilizes the saturation number of overloads followed by the smallest peak loads still available.
3. The procedure of sequencing the blocks in ascending order of peak values available continues after the multiple overload until the minimum growth rate is achieved, then the next available set of multiple overloads is applied (1640 to -340), and so on. It has been calculated, using an equation for the curve of crack growth rate versus distance after the multiple overloads are applied, that only three sets of

multiple overloads are needed to utilize all of the range-mean pair blocks. They are the two sets of 9 mentioned above, and the next block of 18 pairs, 1640 to -160.

A tape is currently being prepared for controlling the fatigue machine to the scheme above.

9.5.5 Semi-Automatic Quantitative Fractography

A heavy involvement with quantitative fractography, which invariably demands the urgent production of crack growth data, has led to the development of a semi-automatic system for the production of fatigue crack growth plots from fracture surfaces in hours rather than days. The system is made up of an optical metallographic microscope fitted with digital micrometer drums and coordinate counters, a BCD interface and multiplexer which supplies the data to a microcomputer (Tektronix 4052) on demand, a magnetic tape storage facility and a digital plotter.

The advantage of this system of data acquisition, reduction, and presentation is the speed with which a routine task may be completed. If the progression markings are readily observable then upwards of 600 data points per hour may be acquired and processed. Most fractures have rather fewer progression markings, and consequently data acquisition, reduction, and plotting of the finished graph may often be accomplished well within the working day. This compares with times measured more in days or weeks using the previous system of manual micrometers, manual data handling and reduction, and manual plotting. Figures 9.2(b), 9.4 and 9.13 have been produced using this system.

9.5.6 NDI Research

Progress in nondestructive inspection research at ARL has continued along the lines described in the previous Review. Theoretical and experimental work on the use of ultrasonic caustics to provide quantitative information about defect size has virtually concluded. It has been shown that caustics can be detected but contrast is very low and reliability is likely to be poor[30]. The successful work on image enhancement[31] has ceased because of higher priority demands.

The in-flight testing[32] of acoustic emission (AE) equipment installed in a jet-trainer aircraft has been completed and laboratory fatigue testing of the aircraft component has commenced. Extensive analysis of the laboratory test data is still needed in order that a confirmed AE - crack growth relationship can be found.

ARL contracted Battelle Northwest Laboratories (USA) to undertake AE monitoring of a full-scale fatigue test of a fighter aircraft with a view to developing in-flight equipment suitable for continuous monitoring and evential development as an airworthiness device. As a result of the advanced nature of the equipment, some testing troubles occurred but these have been largely overcome and an extensive program of data analysis is being undertaken. Support studies have included pattern recognition of spectral signals, some of which forms part of an international cooperative program concerned with the positive identification of signals coming from different sources such as fretting, crack advancement or loading devices.

It has been confirmed that the principal source of AE during crack advancement in several aluminium alloys appears to be the cracking of inclusions[33]. The ARL program on AE as an airworthiness indicator has been described, and the manner in which various topics are interrelated has been shown[34]. Some exploratory work on the location of growing defects by means of AE has also been conducted.

In preparation for undertaking work on NDI of composite materials, an extensive survey of the literature was undertaken, which served to show the deficiencies in many of the presently accepted techniques[35].

Computer controlled NDI equipment has been purchased and will be used to conduct research on the development of advanced NDI techniques. So far, time has been spent on programming the LSI-11 computer and developing software for the ultrasonic and eddy current modular systems.

9.6 BIBLIOGRAPHY ON FATIGUE

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TABLE 9.1

DETECTION AND REPAIR OF SIGNIFICANT CRACKING IN NOMAD AIRCRAFT

LOCATION	Time at Detection (hours)	Length at Detection (mm)	Time at repair (hours)	Length at repair (mm)	COMMENT
Stub-wing rib, actuator cut-out	56 200	15	57 630	15	
Upper wing strut fitting	79 837	undetected	79 837		Critical failure. Fitting replaced.
Stub-wing outboard rib	81 700	40 24 39	103 749	53 24 55	Cracks from 3 holes. Rib replaced.
Main wing front spar web	84 420	10			Current length 49 mm
Stub-wing rib	103 997	22		24	
Stub-wing front spar upper cap	103 997	9			Current length 45 mm
Stub-wing front spar web	105 700	6			Current length 55 mm
Stub-wing front spar lower cap	112 581	undetected	112 581	see Fig. 9.2	Critical failure. Strap repair fitted.

Airframe hours to 31 December 1982: 137 986.

TABLE 9.2

SUMMARY OF MIRAGE FRONT-FLANGE FATIGUE TEST RESULTS

Specimen type	Number of tests	Standard deviation	Log. average life (flights)
A. 5 mm interference-fit bolts	4	0.113	72 524
B. 5 mm clearance-fit bolts	5	0.083	7 695
C. Cold-expanded holes (6.35 mm)	7	0.104	21 570
D. Cold-expanded and interference-fit bush	5	0.055	56 969
E. Interference-fit bushes (8.15 mm)	5	0.139	38 610

The life ratios for the various hole treatments are as follows:

$$\frac{A}{B} = 9.4 \quad ; \quad \frac{A}{C} = 3.4 \quad ; \quad \frac{A}{D} = 1.3 \quad ;$$

$$\frac{A}{E} = 1.9 \quad ; \quad \frac{C}{B} = 2.8 \quad ; \quad \frac{D}{C} = 2.6 \quad ;$$

$$\frac{D}{E} = 1.5 \quad ; \quad \frac{E}{C} = 1.8 \quad .$$

TABLE 9.3

OPERATIONAL DATA RELATING TO FATIGUE METER RECORDS

AIRCRAFT TYPE	OPERATIONAL ROLE	RECORDED TIME		AVERAGES		
		HOURS	LANDINGS	CRUISE ALT. (Ft.)	CRUISE SPEED (Kt.IAS)	WEIGHT (kg)
Shorts 330	Charter Passenger	1792	2238	9500	150	9035 (Av.cruise)
De Havilland DHC-6/100	RPT	2330	3909	Not recorded	145	4880 (Av.cruise)
Embraer EMB 110-P2	RPT	1129	1675 (Approx)	7500	185	5070 (Take-off)
GAF N24	Air Ambulance	263	274	8000	132	3690 (Take-off)

TABLE 9.4

RANGE-MEAN PAIRS FOR AUSTRALIAN MIRAGE WING TEST SEQUENCE, DISCRETIZED TO 14 LEVELS (MICROSTRAIN AT WING ROOT)

Peak	TROUGH												
	-520	-340	-160	20	200	380	560	740	920	1100	1280	1460	1640
-340	0												
-160	0	0											
20	0	0	0										
200	0	0	0	149									
380	1	0	91	9523	2198								
560	1	22	2318	21601	2805	1591							
740	0	37	659	810	1873	1665	705						
920	1	18	131	575	1209	521	896	124					
1100	0	17	72	421	609	302	306	336	110				
1280	3	24	42	400	338	23	131	168	183	147			
1460	5	25	60	516	171	1	92	39	63	71	97		
1640	9	9	18	16	53	0	11	0	0	11	1	35	
1820	1	0	0	0	0	0	0	0	0	0	0	0	0

TABLE 9.5

FATIGUE TEST DATA FOR LOAD CYCLE RECONSTITUTION TESTS

(see section 9.5.4)

Sequence	Retardation	Basic	Acceleration
Programs to failure	14.00 (6 - 14.00*) 12.02	13.54	17.94 (6 - 17.94*) 13.98
Average	13.01		15.96

* The figures in parentheses denote the range of programs over which the crack is clearly identifiable by fractography.

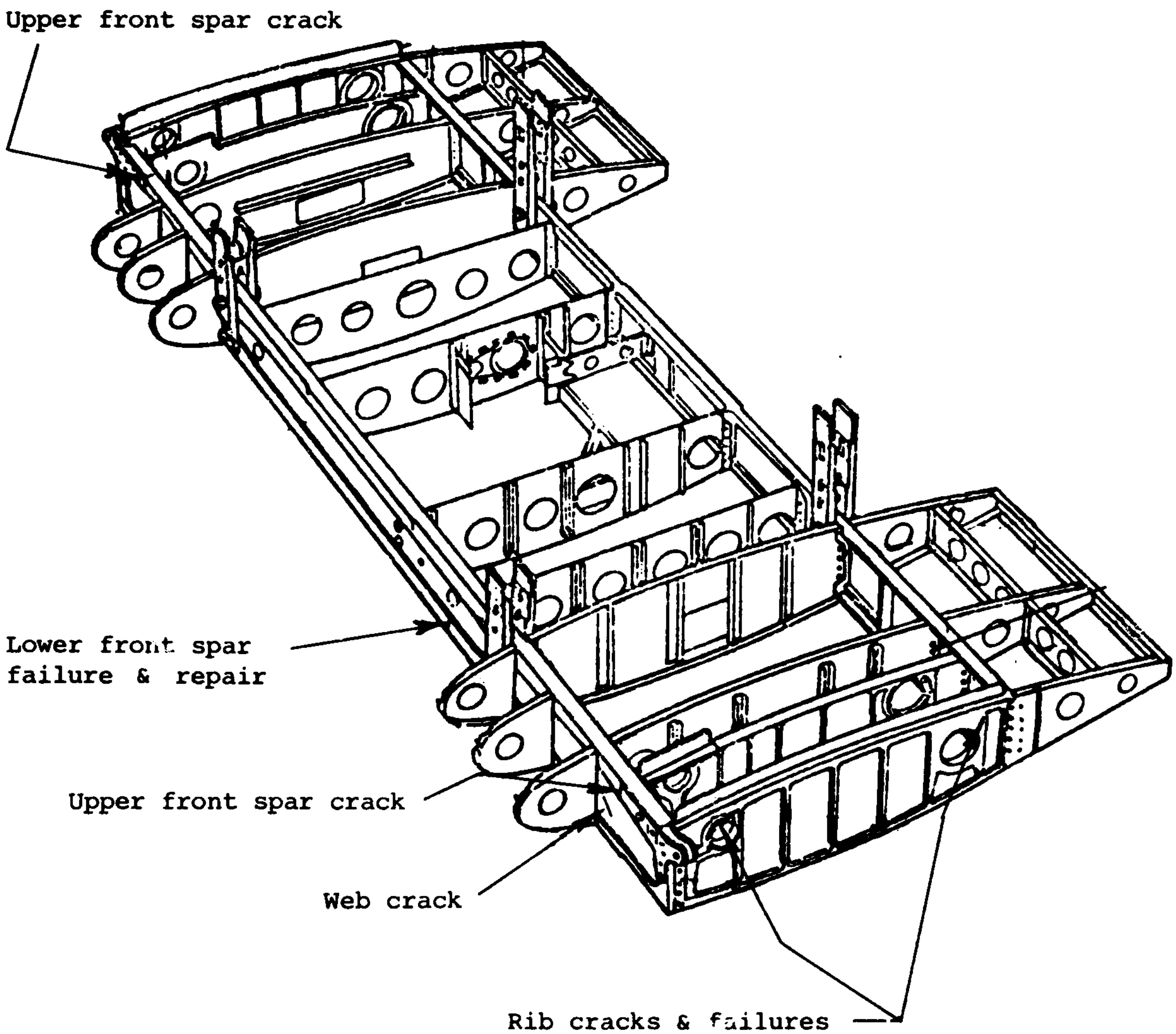


FIG. 9.1 NOMAD STUB WING STRUCTURE SHOWING LOCATIONS OF CRACKING ON FATIGUE TEST

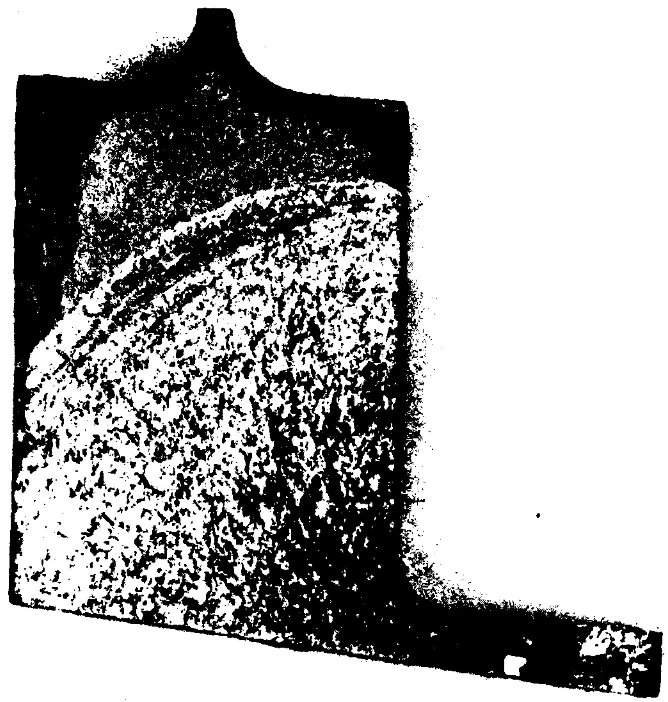


FIG. 9.2(a) NOMAD STUB WING FRONT SPAR FRACTOGRAPH (X2)

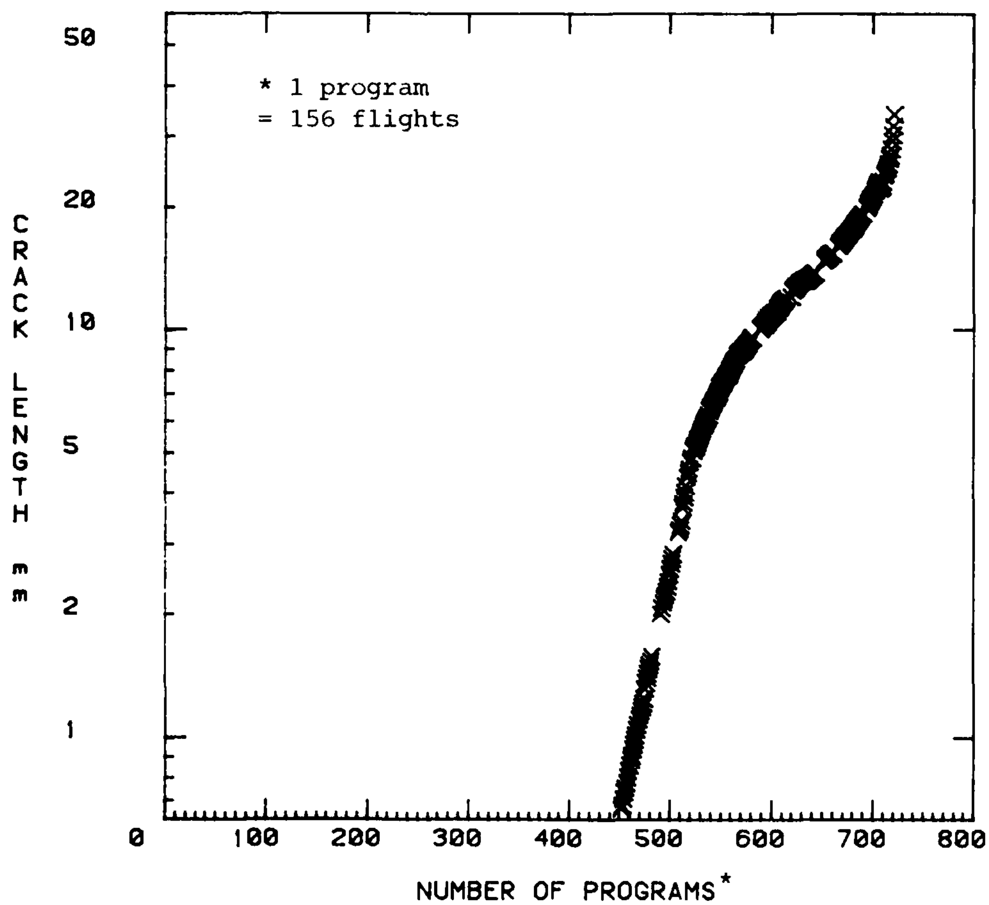


FIG. 9.2(b) NOMAD STUB WING FRONT SPAR CRACK GROWTH

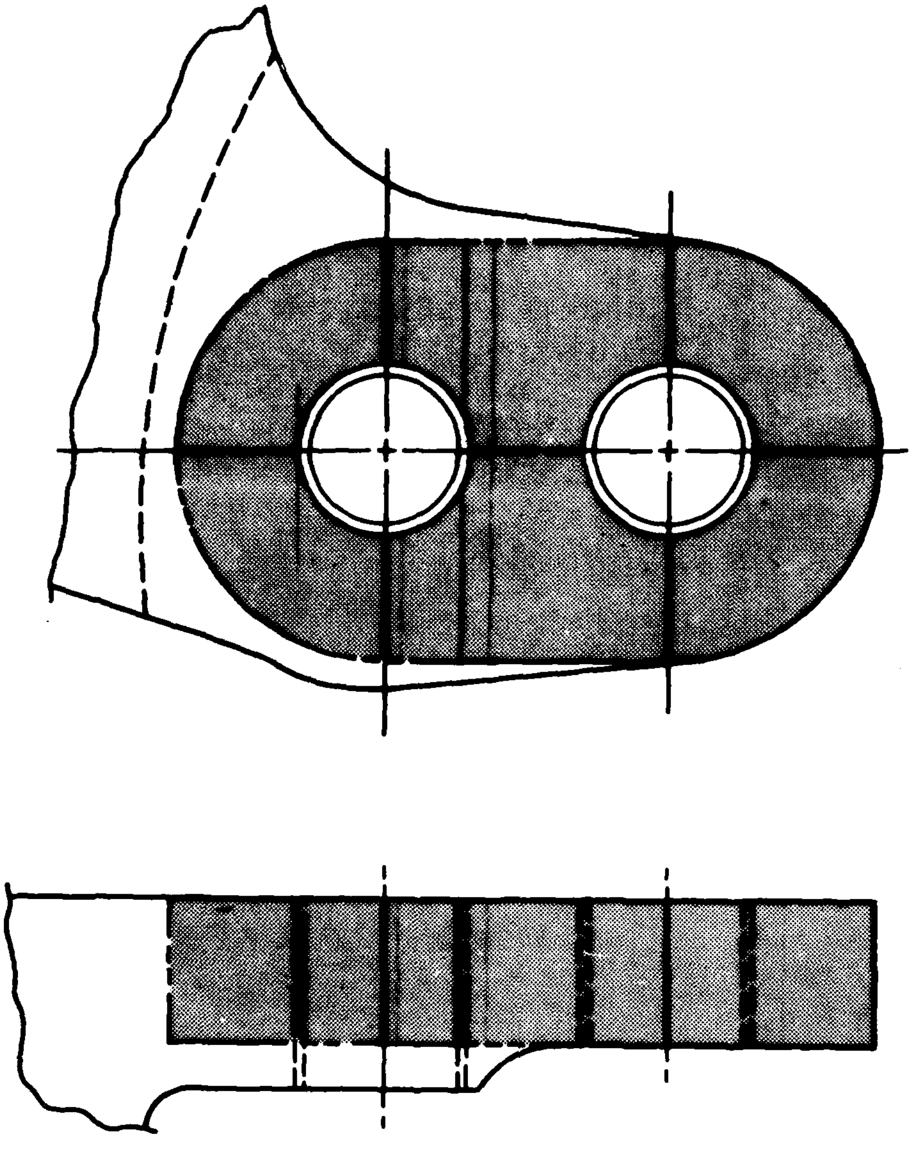
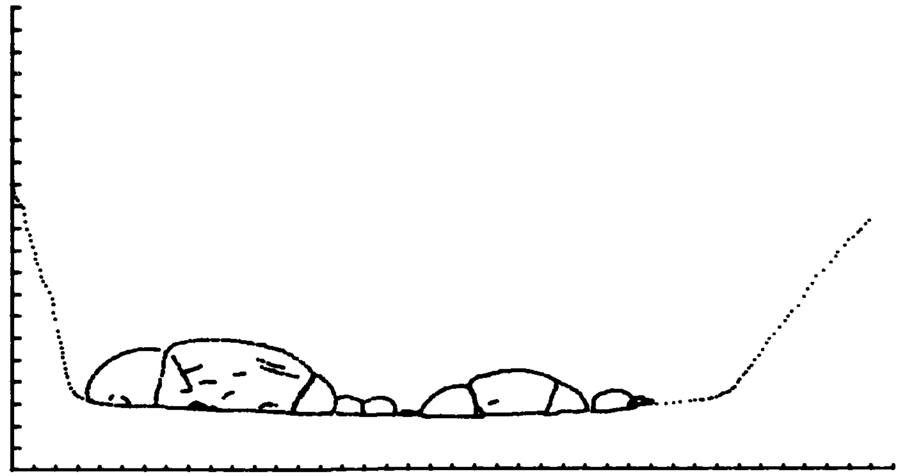
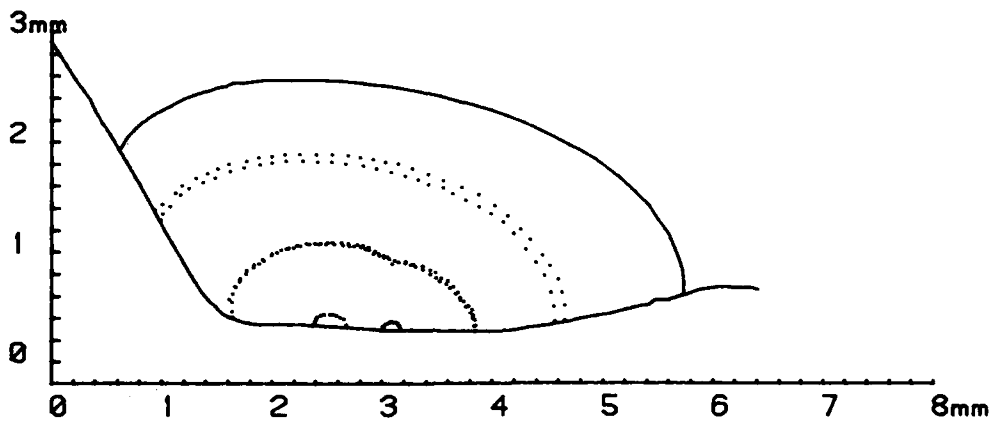


FIG. 9.3 CANBERRA LINK FATIGUE SPECIMEN AND ITS RELATION TO THE CENTRE SECTION FORGING LUG



(a) F-111 A



(b) F-111 C

FIG. 9.4 F-111 WING PIVOT FITTING COLD PROOF TEST FAILURES - CRITICAL CRACK DETAILS

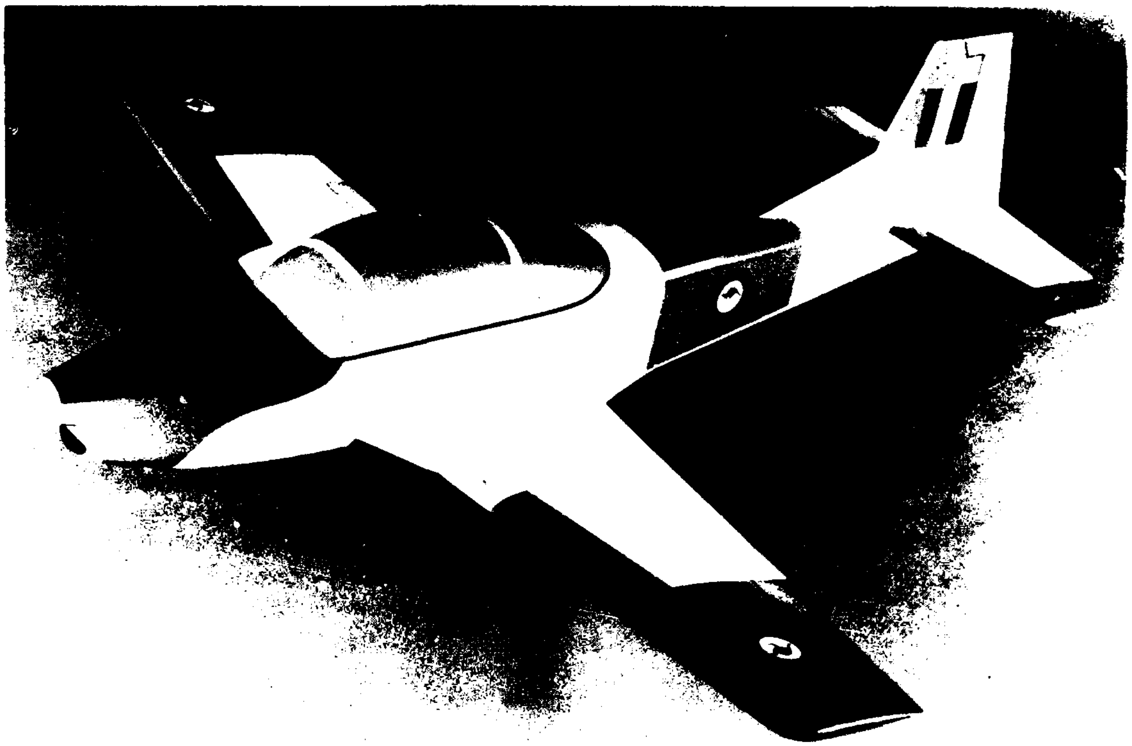


FIG. 9.5 BASIC PILOT TRAINING AIRCRAFT MODEL

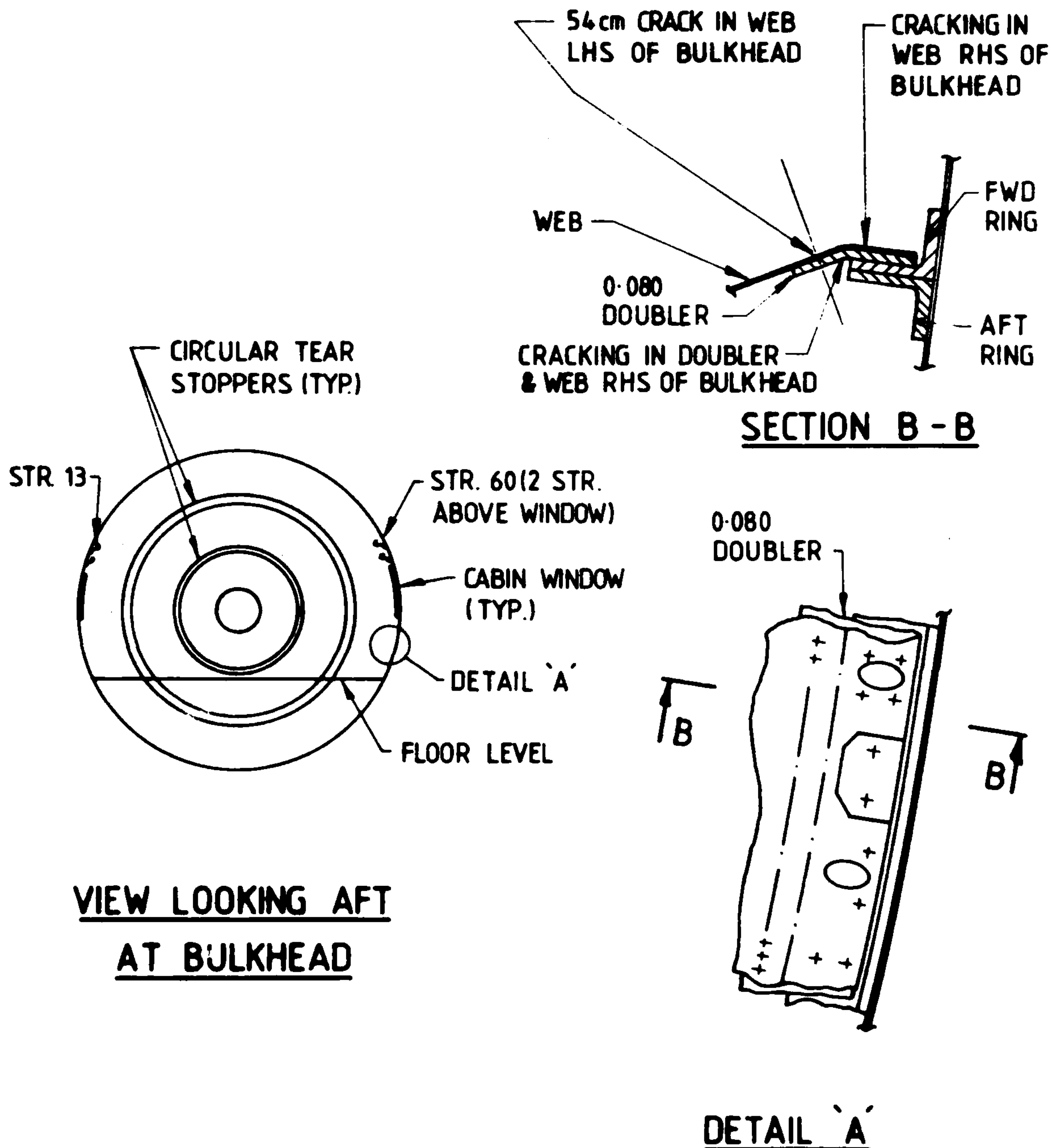


FIG. 9.6 LOCKHEED L188 REAR BULKHEAD DETAIL

FIG. 9.7(a) LOCKHEED REAR BULKHEAD FATIGUE FRACTURES (x0.3)

FIG. 9.7(b) SCANNING ELECTRON MICROGRAPH OF LOCKHEED FATIGUE FRACTURE SHOWING FINE, EVENLY SPACED STRIATIONS (x2300)

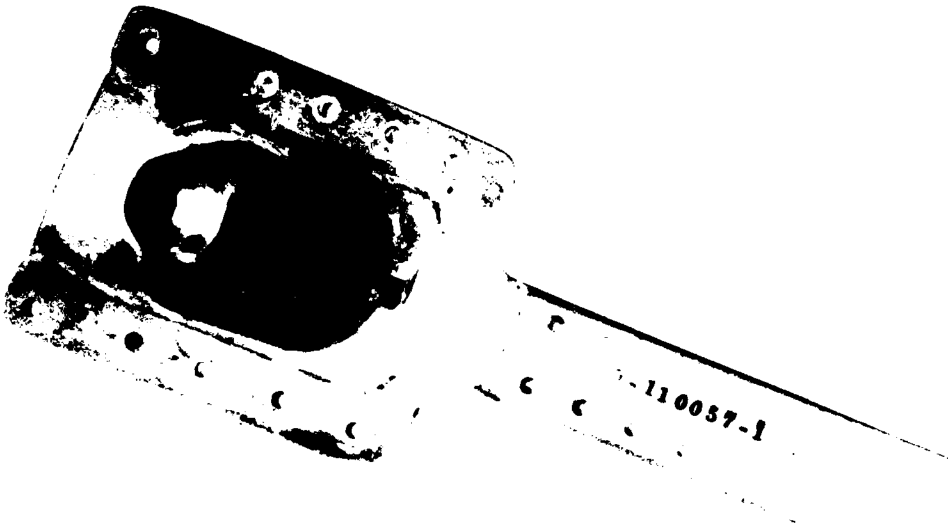


FIG. 9.8(a) OUTERWING ATTACHMENT 'BATHTUB' FITTING FROM BEECH 65-A80

FIG. 9.8(b) BATHTUB FITTING CRACKING REVEALED BY FLUORESCENT PLNTRANT

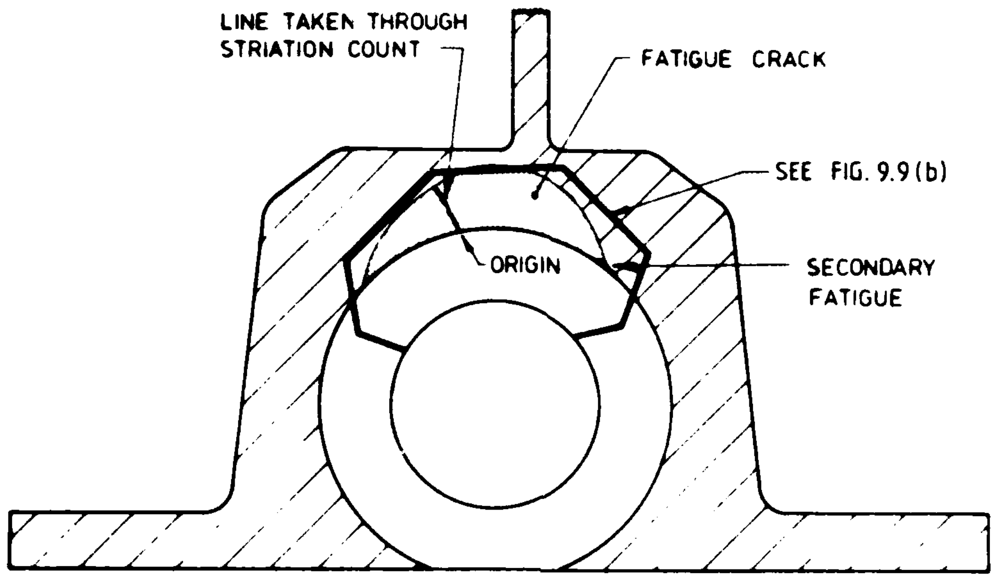


FIG. 9.9(a) CROSS SECTION OF BEECH BATHTUB FITTING CONTAINING FATIGUE CRACK

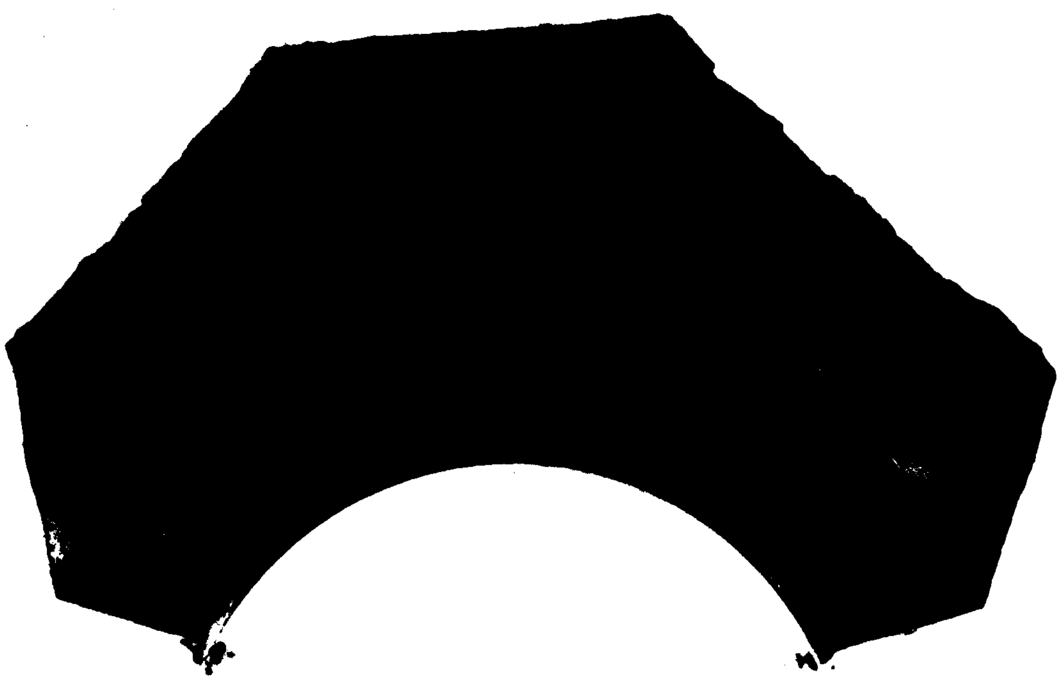


FIG. 9.9(b) SECTION FROM BEECH BATHTUB FITTING SHOWING ORIGIN OF FATIGUE CRACK

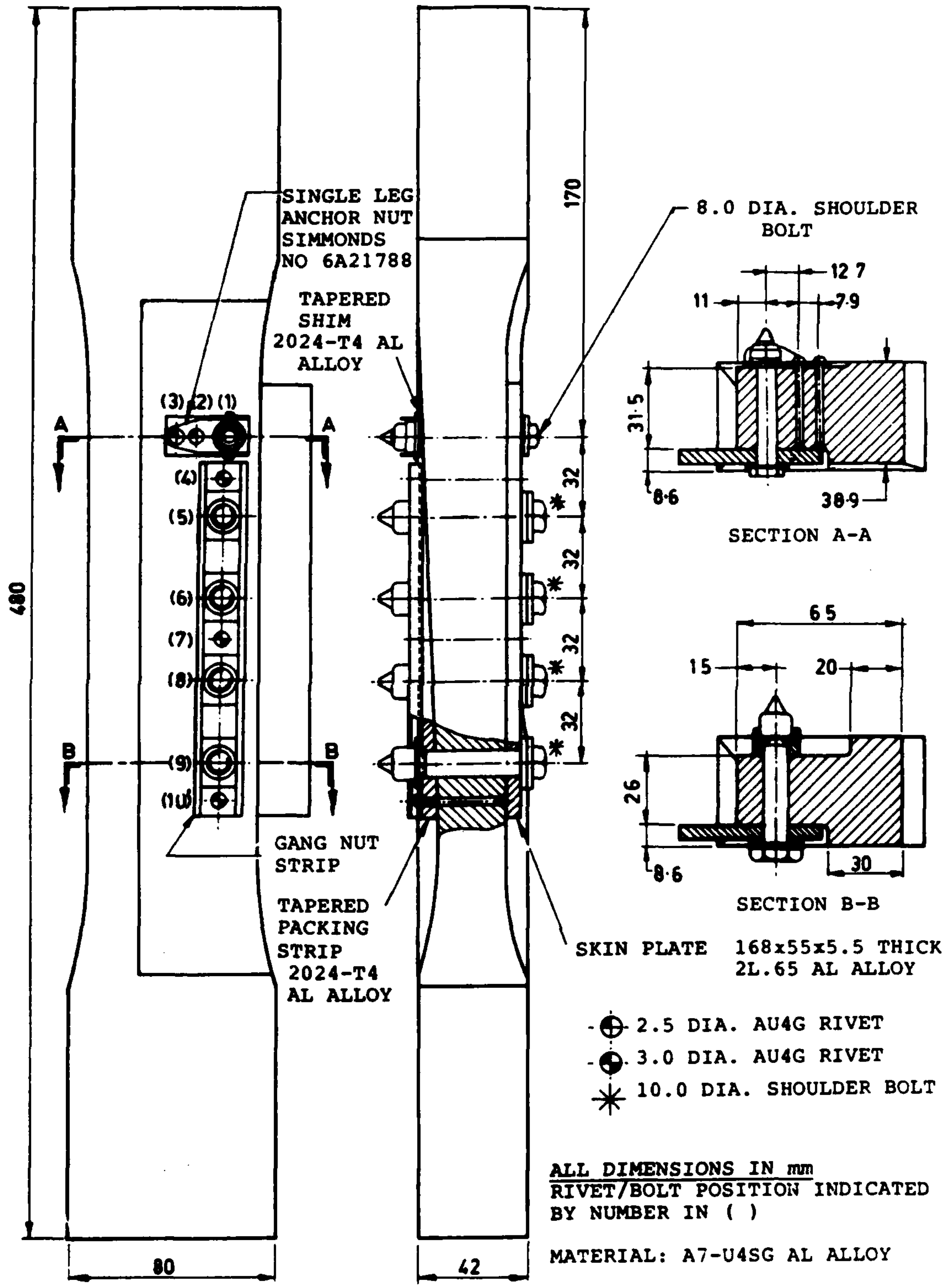


FIG. 9.10 MIRAGE SPAR INNER LOWER REAR FLANGE FATIGUE SPECIMEN

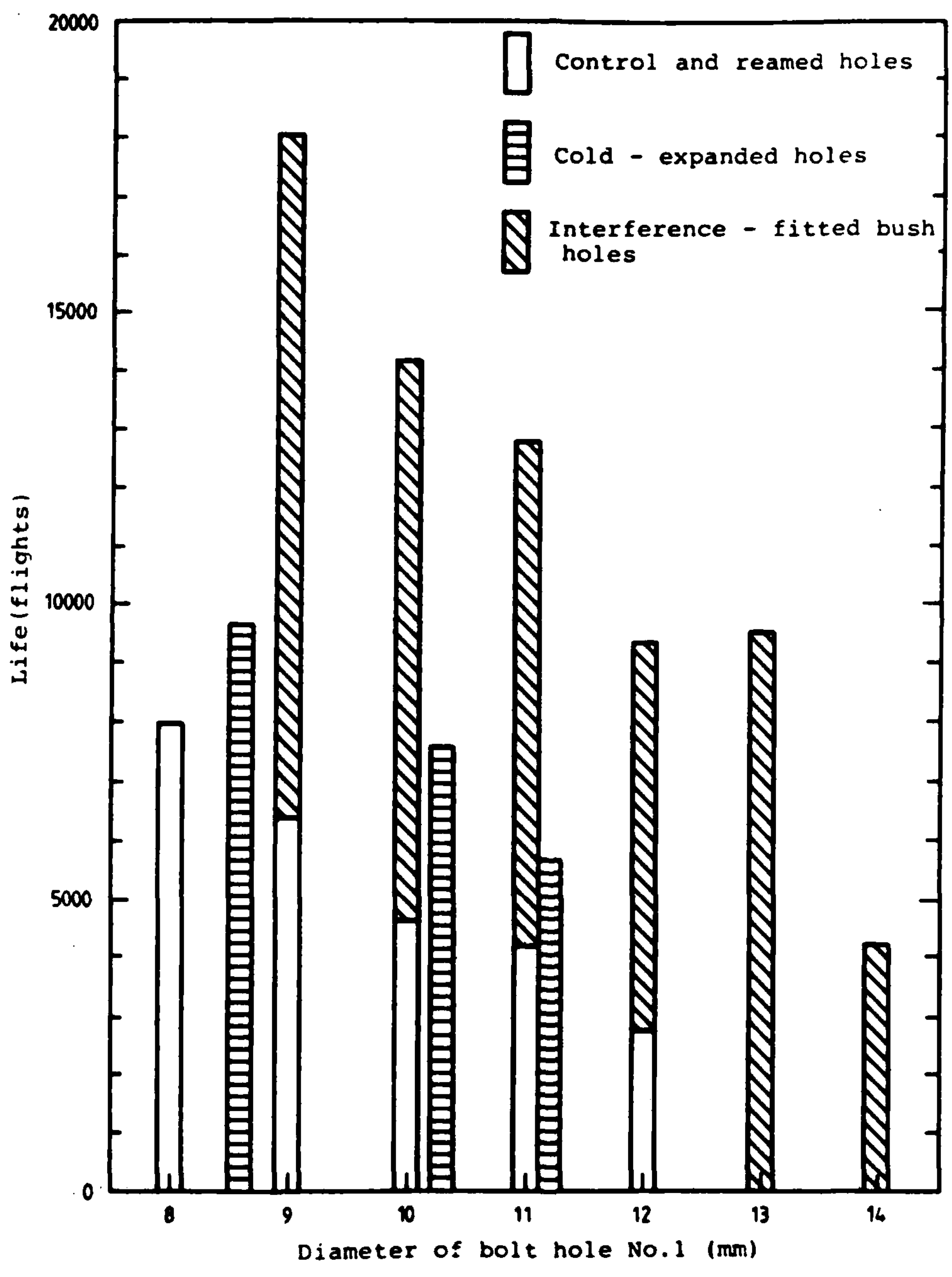
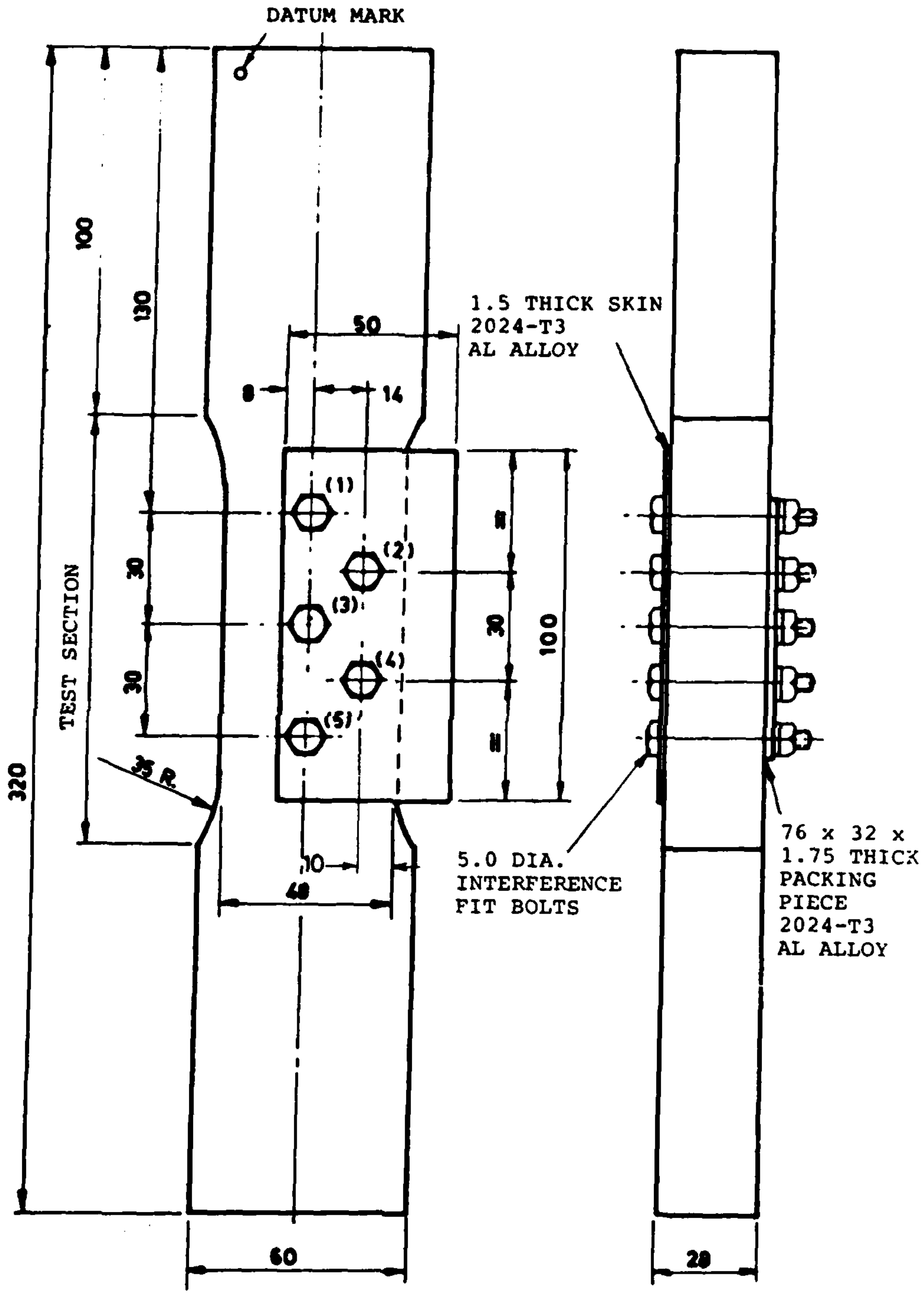


FIG. 9.11 EFFECTIVENESS OF BOLT HOLE REFURBISHMENT TECHNIQUES



ALL DIMENSIONS IN mm
BOLT POSITIONS IN SPECIMEN
INDICATED BY NUMBER IN ( )

MATERIAL: BS.L168 AL ALLOY

FIG. 9.12 MIRAGE SPAR LOWER FRONT FLANGE FATIGUE SPECIMEN

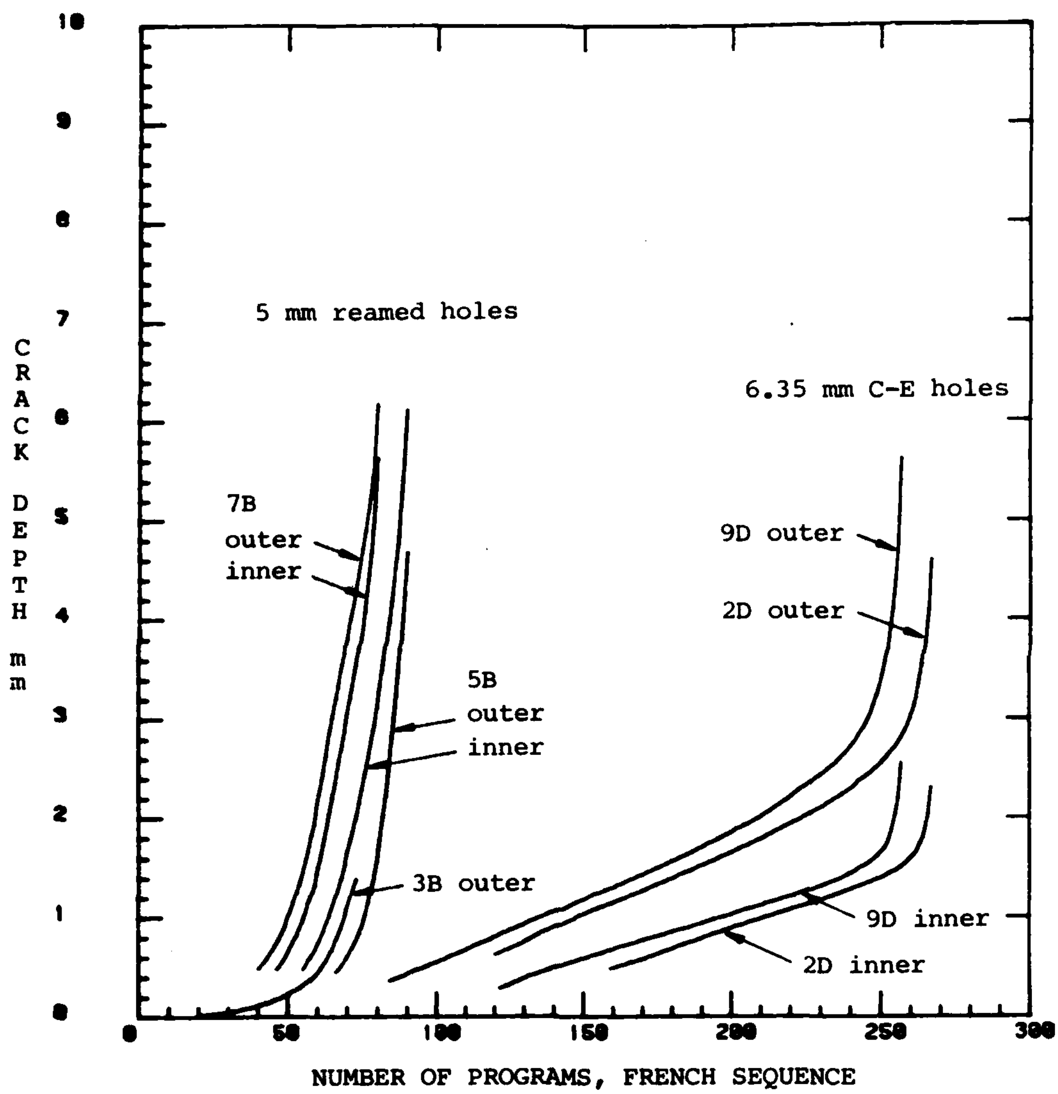


FIG. 9.13 CRACK GROWTH CURVES FOR REAMED HOLES AND COLD-EXPANDED HOLES FROM FRACTOGRAPHIC MEASUREMENTS

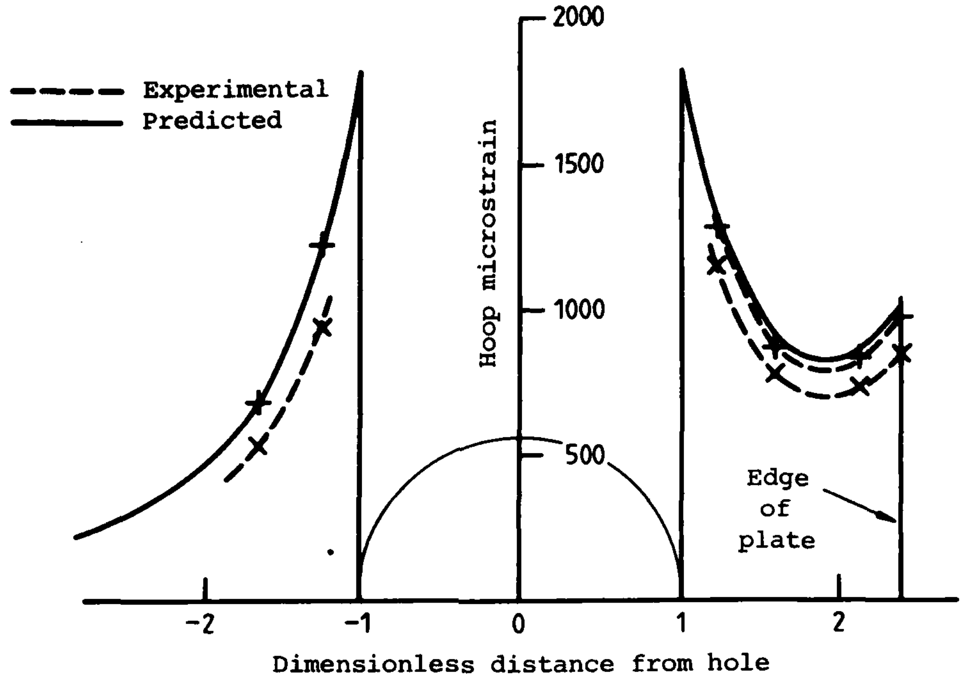
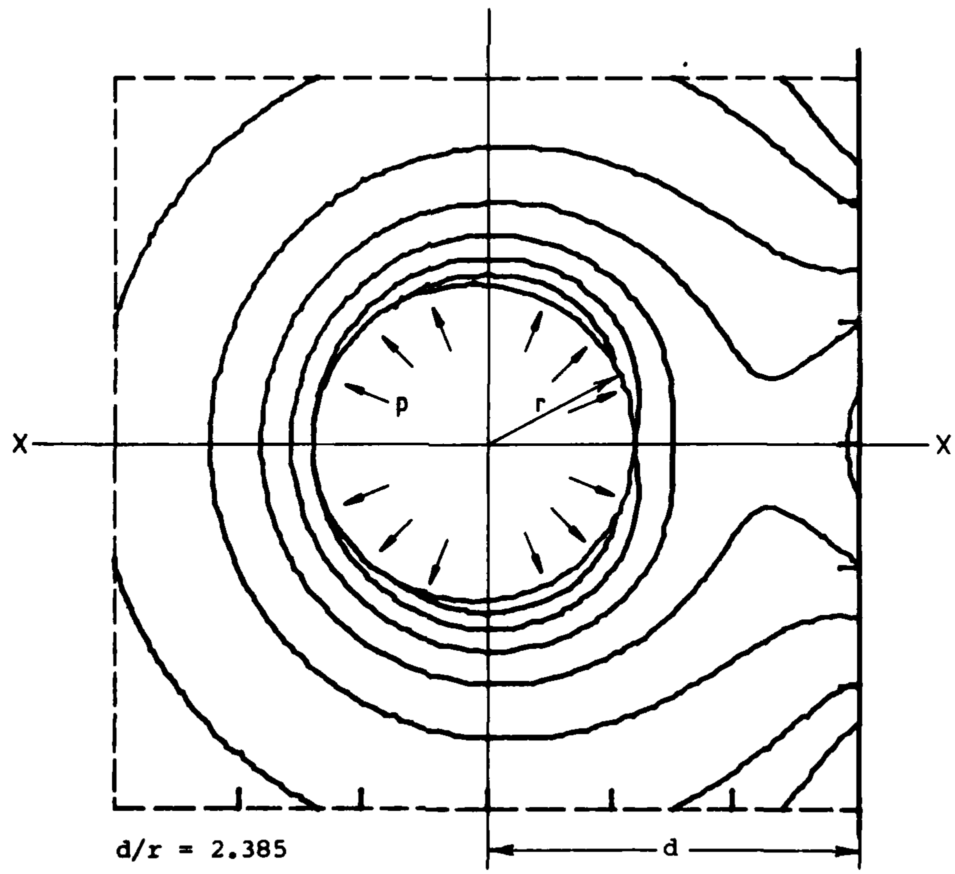


FIG. 9.14 PREDICTED AND MEASURED HOOP STRAINS ALONG XX IN FIG. 9.15



σ_1/p = 1.4 (hole), 1.2, 1.0, 0.8, 0.6, 0.4, 0.2, 0.1

FIG. 9.15 MAJOR PRINCIPAL STRESS CONTOURS FOR PRESSURISED HOLE IN SEMI-INFINITE PLATE

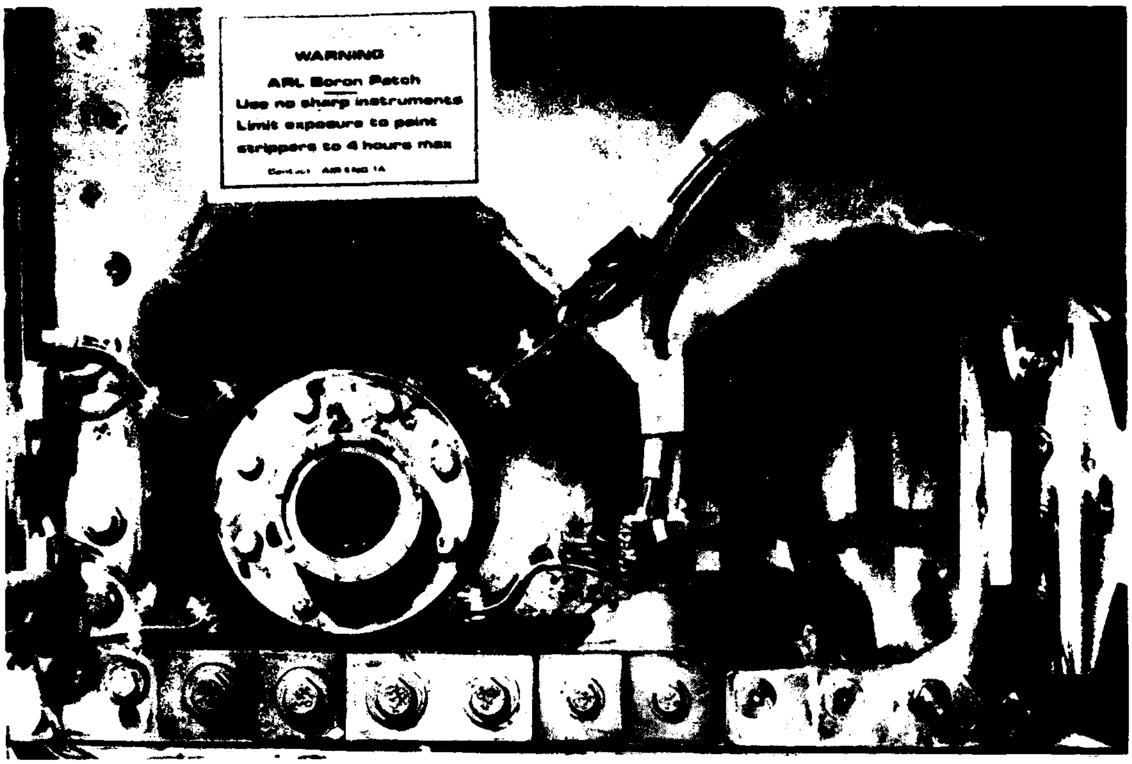


FIG. 9.16 BFRP REPAIRS TO LOWER WING SKIN OF MIRAGE III FATIGUE TEST AIRCRAFT

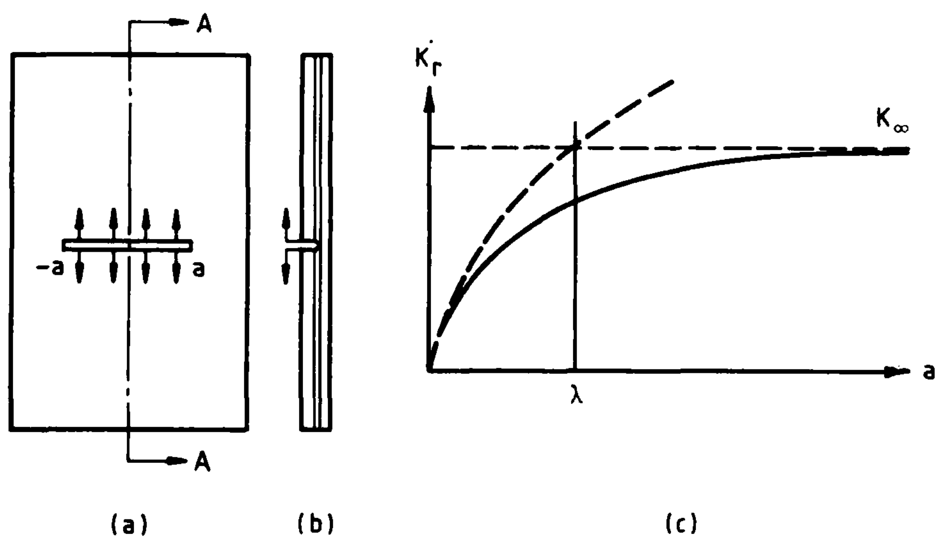


FIG. 9.17(a),(b) CENTRE-CRACKED PLATE BONDED TO AN UNCRACKED REINFORCING PLATE WITH UNIFORM LOADING ON CRACK FACES (c) SCHEMATIC OF STRESS INTENSITY VERSUS CRACK LENGTH

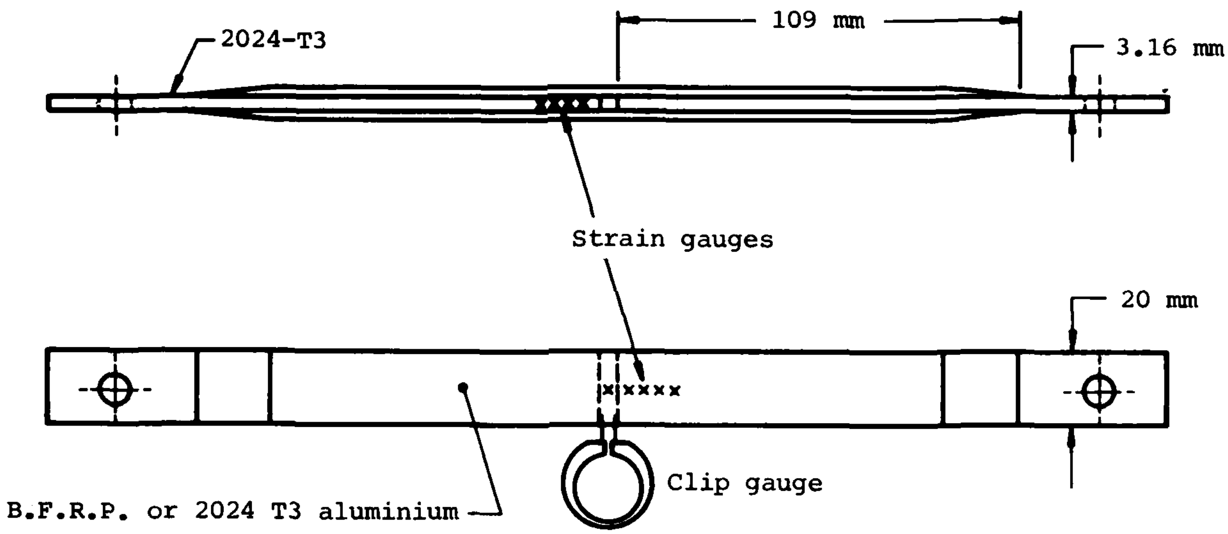


FIG. 9.18 DOUBLE LAP JOINT SHOWING POSITION OF CLIP GAUGE AND STRAIN GAUGES

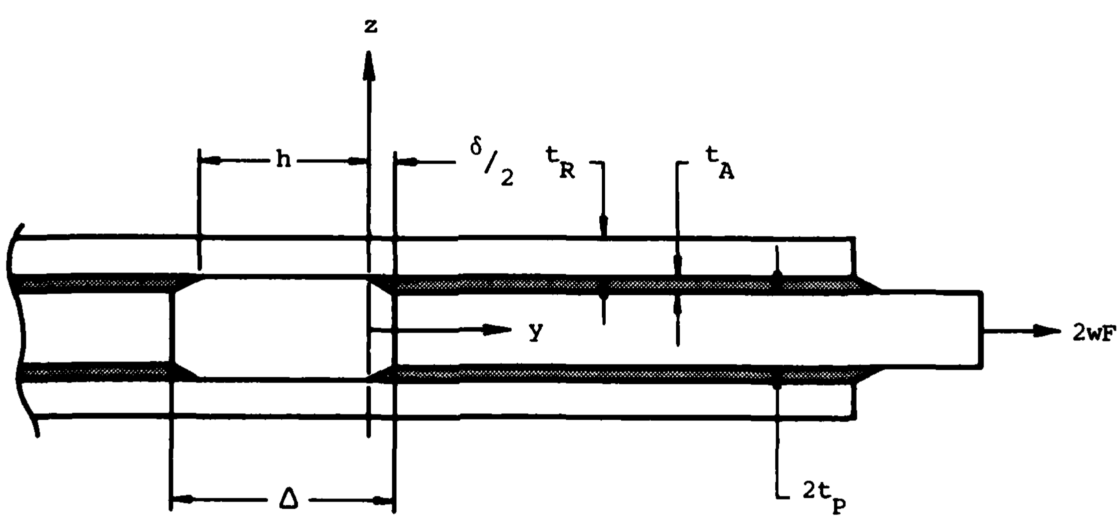


FIG. 9.19 NOTATION USED FOR THICKNESSES AND DISPLACEMENTS

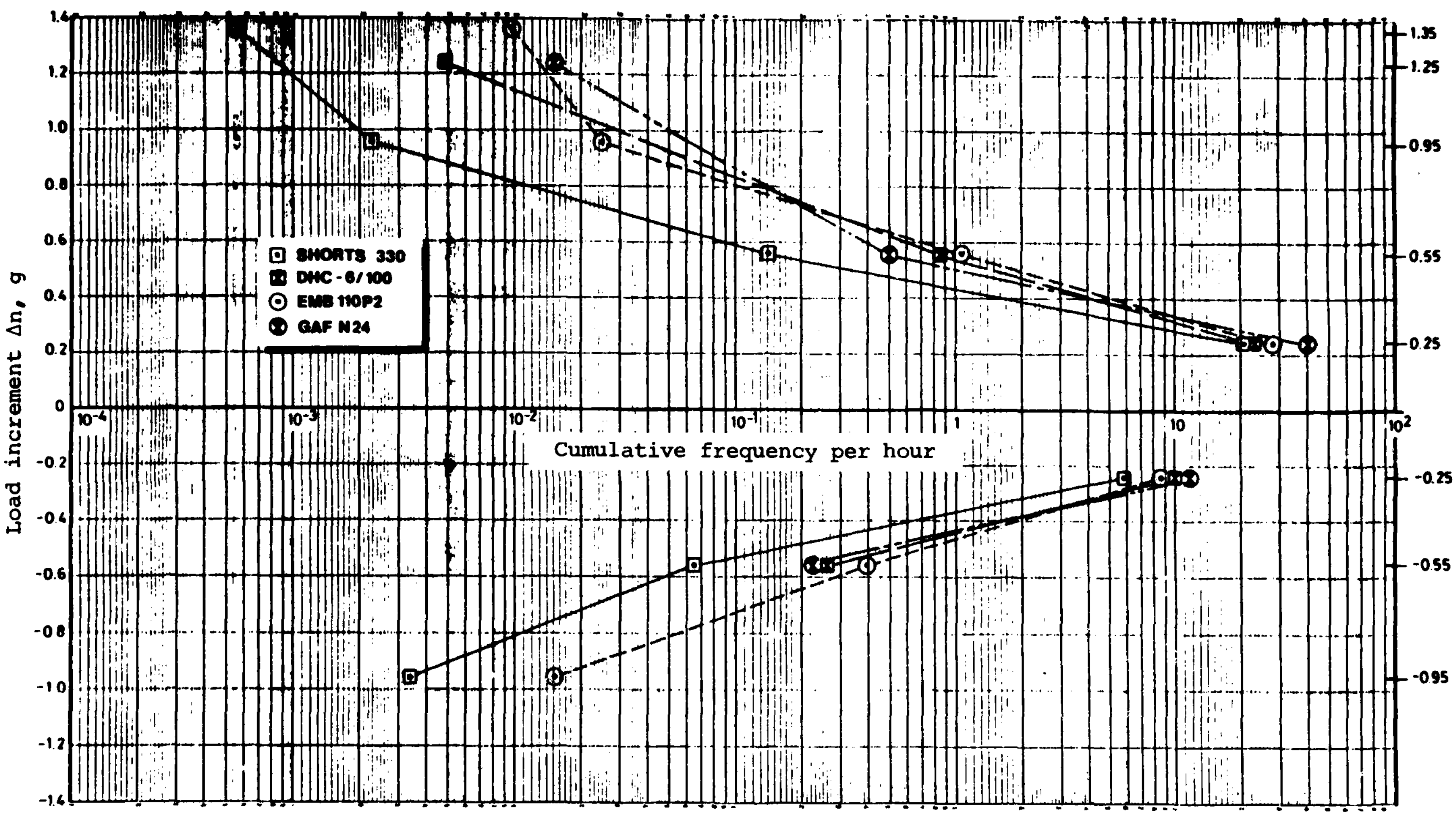
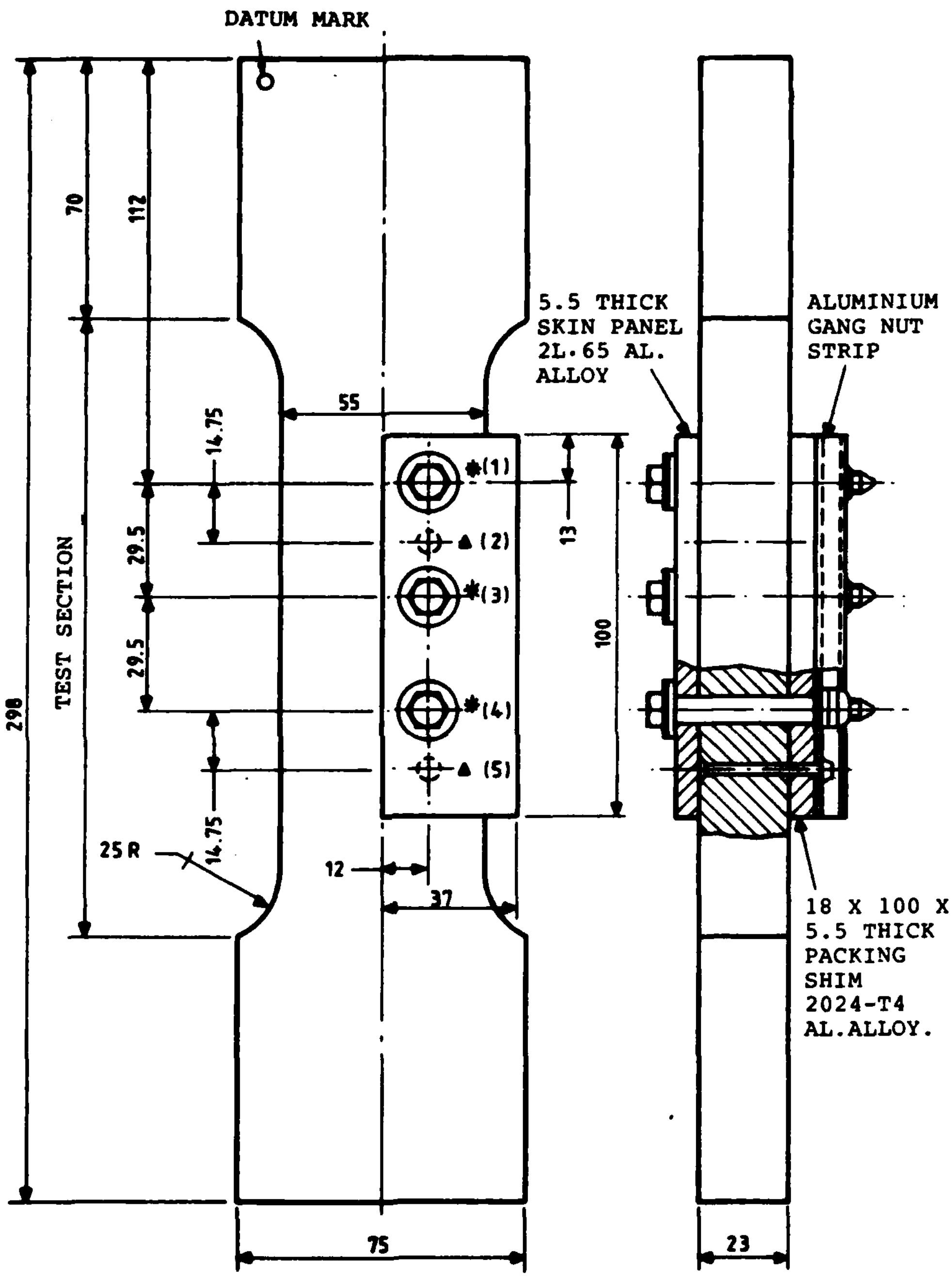


FIG. 9.20 FLIGHT LOADS SPECTRA FOR TWIN TURBOPROP UNPRESSURISED AIRCRAFT



▲ 3.0 DIA. A-U4G RIVET

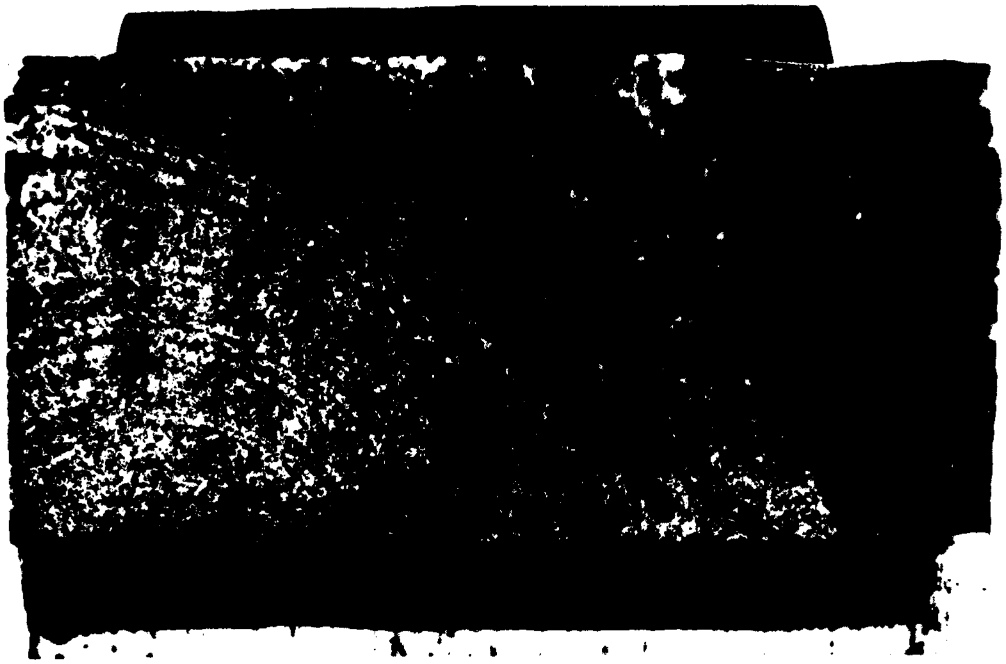
✱ 3.0 DIA. SHOULDER BOLTS (6.0 DIA. THREAD)

RIVET/BOLT POSITION INDICATED
BY NUMBER IN ( )

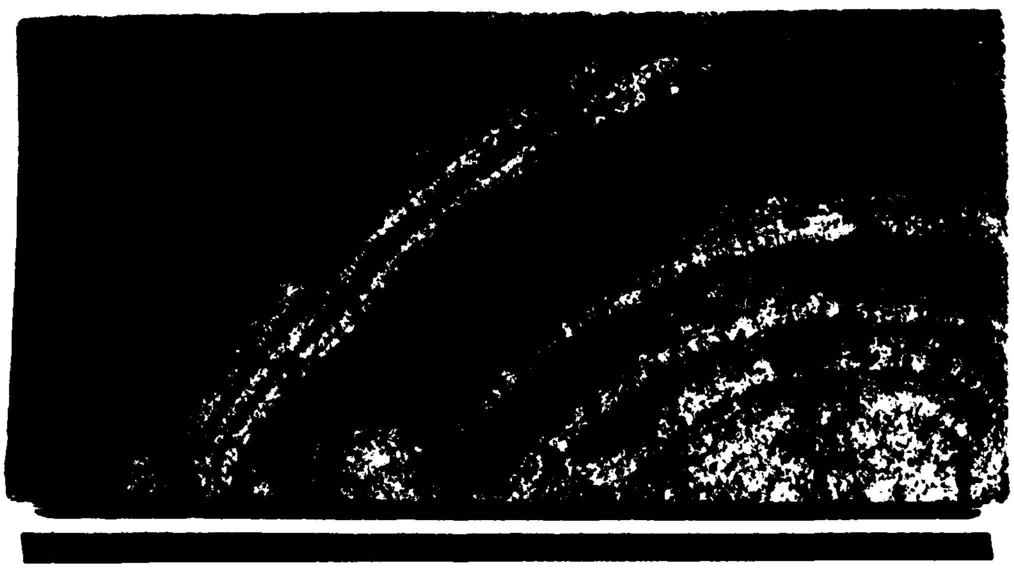
ALL DIMENSIONS IN mm

MATERIAL: 2014-T651
AL. ALLOY

FIG. 9.21 SPECIMEN USED IN LOAD CYCLE RECONSTITUTION STUDY
(MIRAGE SPAR OUTER LOWER REAR FLANGE FATIGUE SPECIMEN)



(a) Retardation sequence (x2)



(b) Acceleration sequence (x2)

FIG. 9.22 FRACTOGRAPHS OF SPECIMENS FATIGUE UNDER SUPPOSED EXTREME SEQUENCES YIELDING IDENTICAL RANGE-PAIR TABLES

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16. Abstract

This document was prepared for presentation to the 18th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Toulouse, France, on May 30 and 31, 1983. A summary is presented of Australian aircraft fatigue research and associated activities. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage repair and refurbishment and fatigue life monitoring and assessment.

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FIG. 9.11 EFFECTIVENESS OF BOLT HOLE REFURBISHMENT TECHNIQUES